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MCR-72-41 (Vol I) Copy No. 99

FINAL REPORT

Space Shuttle Propulsion Systems On-Board Checkout and Monitoring System Development Study (Extension)

VOLUME I

SUMMARY AND TECHNICAL RESULTS

APRIL 1972

Contract NAS8-25619
DRL No. 187, Rev. A
Line Item No. 3 (Issue 2)

Prepared For

George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama

MARTIN MARIETTA
DENVER DIVISION

MCR-72-41

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SPACE SHUTTLE PROPULSION SYSTEMS ON-BOARD CHECKOUT AND MONITORING SYSTEM DEVELOPMENT STUDY (EXTENSION)

VOLUME I - SUMMARY AND TECHNICAL RESULTS

APRIL 1972

Approved by

R. W. VandeKoppel
Program Manager

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MARTIN MARIETTA CORPORATION
Denver Division
Denver, Colorado 80201

FOREWORD

This final report was prepared by the Martin Marietta Corporation under extension to Contract NAS8-25619, "Space Shuttle Propulsion Systems On-Board Checkout and Monitoring System Development Study", for the George C. Marshall Space Flight Center of the National Aeronautics and Space Administration. The report is comprised of two volumes:

Volume I - Summary and Technical Results

Volume II - Guidelines for Incorporation of the On-Board Checkout and Monitoring Function on The Space Shuttle.

CONTENTS

CONTENTS (Continued)

CONTENTS (Concluded)

ABBREVIATIONS

ACPS	Attitude Control Propulsion System
ACT	Acquisition Control and Test
ALRC	Aerojet Liquid Rocket Company
APU	Auxiliary Power Unit
BCU	Bus Control Unit
BIT	Built In Test
BITE	Built In Test Equipment
CBW	Constant Band Width
cc	cubic centimeter
CCC	Central Computer Complex
C&D	Control and Display
C/D	Countdown
CG	Center of Gravity
CMG	Control Monitor Group
COFI	Checkout and Fault Isolation
CSM	Command Service Module
CST	Combined Systems Test
CU	Converter Unit
C&W	Caution and Warning
DCM	Data and Control Management
DIU	Digital Interface Unit
DRS	Data Recording Set
DT	Data Terminal
DTS	Data Transmission System
DTS	Digital Test Set
ECO	Engine Cutoff
EMI	Electromagnetic Interference
EMR	Engine Mixture Ratio
EMV	Electromechanical Valves
EPL	Emergency Power Level
^{o}F	degrees Fahrenheit
FM	Frequency Modulated
FMEA	Failure Mode and Effects Analysis
FO	Fail Operational
FS	Fail Safe
ft^3	cubic feet

ABBREVIATIONS (Continued)

GE	General Electric
GH_2	Gaseous Hydrogen
GIE	Ground Instrumentation Equipment
GN_2	Gaseous Nitrogen
GO_2	Gaseous Oxygen
GSE	Ground Support Equipment
Hz	Hertz (cycles per second)
I.D.	Identification
I/O	Input/Output
IR	Infrared
ISDS	Inadvertent Separation Destruct System
KBPS	Kilobits per Second
K lb	Thousand pounds
KSC	Kennedy Space Center
lb	pound
lb/sec	pounds per second
LCC	Launch Control Complex
LH_2	Liquid Hydrogen
LMSC	Lockheed Missiles & Space Company
LOX	Liquid Oxygen
LPPDC	Launch Pad Power Distribution Control
LRE	Liquid Rocket Engine
LRU	Line Replaceable Unit
LUT	Launch Umbilical Tower
MCF	Maintenance and Checkout Facility
MDC	McDonnell Douglas Corporation
MDS	Main Distribution System
MMC	Martin Marietta Corporation
MPL	Minimum Power Level
N/A	Not Applicable
NAR	North American Rockwell
NASA	National Aeronautics and Space Administration
nmi	Nautical Mile
NPL	Normal Power Level
N_2	Nitrogen
N_2O_4	Nitrogen Tetroxide

ABBREVIATIONS (Continued)

OCMF	Onboard Checkout and Monitoring Function
OCMS	Onboard Checkout and Monitoring System
OITS	Ordnance Item Test Set
OMS	Orbital Maneuvering System
Ox	Oxidizer
Δ P	differential Pressure
PAM	Pulse Amplitude Modulated
PBAN	polybutadiene acrylic acid acrylonitrile
P_c	Chamber Pressure
PCM	Pulse coded Modulated
psia	pounds per square inch, absolute
psid	pounds per square inch differential
PSV	Pressure Sequencing Valve
PSVOR	Pressure Sequencing Valve Override
PTPCS	Propellant Transfer and Pressurization Control Set
PU	Propellant Utilization
^oR	degrees Rankine
RF	Radio Frequency
RIGS	Resonant Infrasonic Gaging System
RMIS	Remote Multiplex Instrumentation System
RMU	Remote Multiplexer Unit
RPM	Revolutions per Minute
S/A	Safe/Arm
sccs	Standard cubic centimeters per second
SCI	Spacecraft Electronics Incorporated
SFC	Squib Firing Circuit
SOCVU	Standard Ordnance Circuit Verification Unit
SPS	Service Propulsion System
sps	samples per second
SRM	Solid Rocket Motor
SSME	Space Shuttle Main Engine
std	Standard
SVD	Stray Voltage Detector
TBD	To Be Determined
TCPS	Thrust Chamber Pressure Switch
T&FS	Tracking and Flight Safety
TPA	Turbopump Assembly
TPS	Transient Power Supply
TVC	Thrust Vector Control

ABBREVIATIONS (Concluded)

UDMH	Unsymmetrical Dimethylhydrazine
UV	Ultraviolet
VAB	Vertical Assembly Building
VDC	Volts Direct Current
VECOS	Vehicle Checkout Set
VIB	Vertical Integration Building
VPDC	Van Power Distribution and Control
wk	Week

SUMMARY

This study was conducted as an extension to basic Contract NAS8-25619. The Phase B system and main engine studies and certain related technology studies were reviewed under Task 1 to update the results of the basic study. The resultant conclusions and recommendations included the following:

> Techniques exist for bearing incipient failure detection, but require further development. No singularized approach for leak detection and monitoring appears feasible; several potential techniques for particular leak detection applications were recommended for further development.

Task 2 of the study consisted of generating a guidelines document that delineates the approach, methodology and verification requirements for defining and analyzing the propulsion systems to establish and implement the checkout and monitoring requirements. The resultant document, Guidelines for Incorporation of the Onboard Checkout and Monitoring Function on the Space Shuttle, is published as Volume II of this report.

Task 3 was conducted to define the checkout and monitoring requirements of the Titan III L Space Shuttle booster, and to define an approach for implementing the requirements. Conclusions and recommendations of Task 3 are summarized as follows:

> The onboard checkout and monitoring functions defined in the Guidelines For Incorporation document are applicable to an expendable booster; the degree of onboard capability incorporation differed between the baseline expendable booster configuration and the recoverable, reusable configuration evaluated in the basic study.
>
> The booster checkout and monitoring impact on orbiter requirements was slight, as was the impact on existing propulsion hardware design.
>
> The booster propulsion systems' measurement requirements, as derived in the study, consisted of 35 control measurements, 28 fault detection measurements and 125 performance analysis measurements, for a total of 188 propulsion system measurements.
>
> Areas recommended for futher technology work to enable optimum incorporation of the checkout and monitoring function consisted of a solid rocket motor case burnthrough detector, and leakage detectors.

I INTRODUCTION

I INTRODUCTION

A. BACKGROUND

Under basic Contract NAS8-25619, the Martin Marietta Corporation analyzed the Space Shuttle propulsion systems to define the onboard checkout and monitoring function. A baseline Space Shuttle vehicle and mission was used to establish the techniques and approach for defining the propulsion systems' checkout and monitoring requirements, and for analyzing these requirements to formulate criteria for implementing the functions of preflight checkout, performance monitoring, fault isolation, emergency detection, display, data storage, postflight evaluation, and maintenance retest. The final report on this basic contract was completed and distributed in April, 1971.

In May, 1971, work was initiated on an extension to the basic contract. Space Shuttle systems and technology studies had been conducted concurrently with the basic contract; such studies that included elements pertinent to the checkout and monitoring function were identified for review to update and expand the results of the basic study. This review and updating comprised Task 1 of the program extension. Under Task 2 of the extension, the design guidelines and constraints, concept development procedures, and general methodology developed in the basic study, updated to utilize the results of the concurrent Space Shuttle studies, formed the basis for generation of guidelines for the incorporation of the checkout and monitoring function. Also, because expendable boosters for the Space Shuttle vehicle came under consideration after initiation of this contract extension, Task 3 was added to evaluate the propulsion checkout and monitoring requirements and implementation criteria for a selected expendable booster.

This final report presents the results of the three tasks performed under the contract extension. The report is comprised of two volumes. Volume I describes the objectives and approach for each of the three tasks, and presents the results of Task 1 and Task 3. Volume II contains the Task 2 results, i.e., "Guidelines for Incorporation of the Onboard Checkout and Monitoring Function of the Space Shuttle".

B. APPROACH

The approach used in the study is illustrated in Figure I-1. Tasks 1 and 2 were conducted concurrently, with Task 3 being initiated at the midpoint of the program. Monthly and quarterly progress reports were issued, and formal program reviews were conducted at approximately six-week intervals.

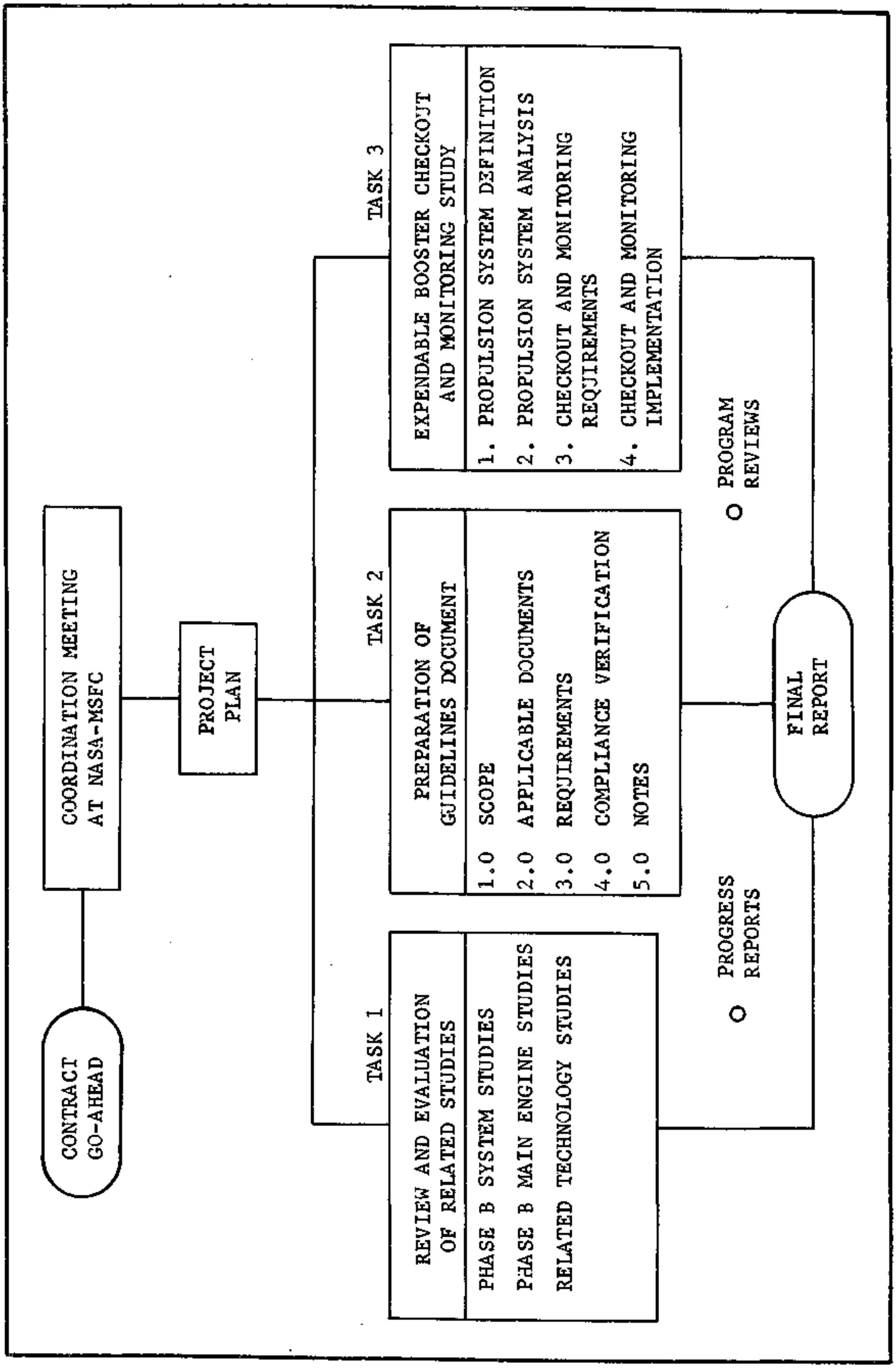

Figure I-1 Study Approach

1. Scope: Task 1 was comprised of updating the basic study by reviewing results of the Space Shuttle System Phase B Studies, the Space Shuttle Main Engine Phase B Studies, and selected related Space Shuttle technology studies. The following studies were subjected to review in this task:

   Space Shuttle System Phase B Studies - Contracts NAS8-26016 and NAS9-10960 by McDonnell Douglas and North American Rockwell were vehicle definition studies that addressed all phases of operation and defined system design approaches. Our review of these study results was concentrated on the orbiter.

   Space Shuttle Main Engine Phase B Studies - Space Shuttle main engine definition studies were conducted under Contracts NAS8-26186, -26187 and -26188 by Pratt & Whitney, Rocketdyne, and Aerojet, respectively. The Rocketdyne study results were reviewed; because of the unavailability of documentation on the other two contracts, the effort originally allocated for their review was reallocated to the special interest areas of bearing incipient failure detection and leakage detection.

   Related Technology Studies - The results of the following related studies were assessed in this task:

   Contracts NAS10-7145 and NAS10-7788 (General Electric). The basic study was an evaluation of techniques for automatic self-contained readiness assessment of mechanical components. The follow-on program is an evaluation of structure borne acoustics as a readiness assessment technique.

   Contract NAS10-7291 (Pearce and Associates). The objectives of this study were to define a faster and more reliable method to detect tank and system leaks, to demonstrate a prototype leak detector, and to develop associated flight weight electronics.

   Contract NAS9-11330 (Lockheed Missiles and Space Company). This study defined and evaluated cryogenic supply subsystems for the orbiter.

Contract NAS8-26378 (SCI Electronics, Inc.). The objective of this study is to define the optimum data bus system for Space Shuttle.

Contract NAS10-7258 (Martin Marietta Corporation). This study of propellants and gases handling for Space Shuttle included an assessment of techniques for propellant level sensing and quantity gaging.

The specific approach used in evaluating these study results are presented in Chapter II of this volume, together with the results of the task.

Task 2, consisting of the generation of guidelines for the incorporation of the checkout and monitoring function, was conducted by incremental preparation, submittal and review of each of the sections of the document. Chapter III of this volume describes the task objectives and results, and the resultant guidelines document forms Volume II of this report.

Task 3 was conducted to define the checkout and monitoring function of the propulsion systems of an expendable booster for Space Shuttle. Titan III L was selected as the baseline booster for the study. The approach followed was derived in the basic study and consisted of defining in detail the propulsion and associated avionics systems, analyzing the propulsion systems to establish control, checkout and monitoring requirements, and conducting further analysis to define the techniques for implementing the checkout and monitoring function. The results of this task are presented in Chapter IV of this volume.

C. PROGRAM PERSONNEL

The following personnel made primary contributions to this program:

Martin Marietta Corporation

D. Del Duca	A. O'Connell
R. Drew	R. VandeKoppel (Program Manager)
R. Mueller	R. Whitted

NASA

J. Fries - MSC
D. Moja - KSC
S. Tulloch - GE/HVO
H. Harman - MSFC
E. Mintz - MSFC
B. Nunnelley - MSFC
D. Thompson - MSFC
W. Voss (COR) - MSFC

Additionally, we gratefully acknowledge the support and contributions of numerous other NASA, Martin Marietta Corporation, General Electric Company, Lockheed Missiles and Space Company, McDonnell Douglas Corporation, Pearce and Associates, Rocketdyne, and SKF Industries, Inc. personnel in the conduct of this study. Specific individuals who supplied technology information on bearing incipient failure detection and leakage detection are identified in the text of the appropriate paragraphs.

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II TASK 1 - RELATED STUDIES

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A. TASK 1 OBJECTIVES

1. Scope - Task 1 consisted of updating the findings of the basic study through review and evaluation of the Space Shuttle Phase B System Studies, the Phase B Main Engine Studies, and certain other related technology studies.

   a. Phase B System Studies - Contracts NAS9-10960 and NAS8-26016 by North American Rockwell and McDonnell Douglas, respectively, were Space Shuttle vehicle definition studies that addressed vehicle configuration definition and operations.

   b. Phase B Main Engine Studies - Space Shuttle main engine definition studies were conducted by Pratt & Whitney, Rocketdyne, and Aerojet under Contracts NAS8-26186, NAS8-26187, NAS8-26188, respectively.

   c. Related Technology Studies - Contract NAS8-26378 is a study of data bus techniques for Space Shuttle. It is being conducted by SCI Electronics, Inc.

      Contract NAS9-11330 is a study that is being conducted by Lockheed Missiles and Space Company. Its purpose is to define and optimize techniques for the cryogenic fluid storage and supply functions of the Space Shuttle.

      Contract NAS10-7145 was a study conducted by General Electric to define and evaulate techniques for automatic self-contained readiness assessment and fault isolation of ground and on-board mechanical systems. A follow-on study under Contract NAS10-7788 is being conducted by General Electric to evaluate structure-borne acoustics as a technique for mechanical systems readiness assessment.

      NAS10-7291 (Pearce Associates): This study was conducted to develop improved methods to detect leakage in fluid systems for the Space Shuttle. A portion of the documentation from this study was previously reviewed.

      Contract NAS10-7258 (Martin Marietta Corporation) was a study of propellants and gases handling in support of Space Shuttle. The study included a task that reviewed propellant level sensing and quantity gaging techniques.

2. Approach - Because of the basic differences in content and objectives of these related studies, a checklist system was employed for the review and evaluation. The topics comprizing the checklist are described below:

a. Major Propulsion System Differences - This item pertains to the identification of differences between the propulsion systems of the Design Reference Model of the basic study and those of the related Phase B studies. Of interest were those differences that would affect the results of the basic study. Emphasis was placed on those differences that directly affect subsequent items of this checklist.

Candidates of potential significance in this regard included the following:

Main engines and main engine controllers;

Airbreathing engines and engine controllers;

Auxiliary Propulsion System assemblies;

Orbital Maneuvering System;

Assemblies and components such as valve actuators, regulators, check valves, propellant gaging systems, and auxiliary power unit drive assemblies.

b. Parameter Definition and Measurement Selection - Parameter definition is the identification and specification of the physical quantity about which certain information is required to accomplish the objectives of the checkout and monitoring function. Measurement selection is the method employed by the checkout and monitoring function to acquire the desired information. The basic steps necessary to arrive at comprehensive parameter definitions, which in turn lead to measurement selections, are the failure modes and effects analysis and the checkout and monitoring requirements analysis. The review included an assessment of these analyses for their adequacy in making the specified parameter definitions and measurement selections. Attention was given to eliminating duplicate and nonessential parameters, in addition to identifying any simplifying steps which the Phase B study teams devised in arriving at their parameter definitions and measurement selections.

c. Sensing Methods, Sensing Selection, Sensor Technology Status and New Requirements - An evaluation was made of the sensors identified by the related studies to establish whether solutions were derived in those areas that the basic NAS8-25619 defined as potential new technology requirements.

d. <u>Sensor Characteristics, Sensor Location, Sample Rate, and Built-in Test Equipment Requirements</u> - In conjunction with the proceeding review item, the sensor specifications of the related and basic studies were evaluated to ensure that minimum requirements consistent with the intended application were specified.

e. <u>Special Sensing vs. Use of Functional Signals</u> - The sensing techniques defined in the various studies were evaluated to determine whether those parameters or types of parameters for which functional inputs and outputs could be used as a source of data were recommended in lieu of using a special or dedicated sensing device to acquire the requisite information. This effort included an assessment of the criticality assigned to the various propulsion elements along with the corresponding recommendations for failure detection to determine whether or not some or all of the sensors assigned to an element could be eliminated through the use of existing functional signals.

f. <u>Checkout and Monitoring vs. Statistical Replacement</u> - Statistical replacement is the use of operating data, in the form of running time or number of cycles or events which an element has accumulated or undergone in its lifetime, to determine when it is due for maintenance or replacement. This technique is based on the use of extensive test and operating histories of a particular element and a statistical prediction of when such elements are likely to exhibit high failure rates. The applicable material was reviewed to determine where statistical replacement had been specified, and, in such cases, to identify the rationale that led to this recommendation.

g. <u>Degree of Preflight, Inflight, and Postflight C/O&M Usage</u> - A set of criteria and concepts was developed during the basic study from which the functions of onboard checkout and monitoring of the Space Shuttle propulsion systems during preflight, inflight, and postflight were delineated. The Phase B studies were examined to identify departures from the delineation, along with supporting rationale.

h. Significance of Data - The significance attached to the data being collected and processed, in terms of data usage, was reviewed and compared with the basic study results. The task identified data to be recorded for future trouble-shooting and refurbishment or trend analysis displayed to the crew for status or caution and warning, and data to be discarded because it indicates normal propulsion system operation.

i. Onboard Checkout vs. Ground Support Equipment - The basic study recommended a small amount of ground equipment for checkout and monitoring purposes. The related Phase B study results were analyzed to determine whether or not they contained improved techniques for performing the functions for which ground equipment was identified, and whether some of the ground equipment was replaced by on-board capability.

j. Impact of Requirements of Other Systems - This item deals with the impact of simultaneous demands of the other onboard systems on the onboard capabilities of the vehicle data bus, the central computer processing capability, the data storage facilities, the caution and warning systems, the display facilities, and the crew. The related studies were evaluated for information relating to the impact of other systems on common onboard resources to determine a potential degradation of performance of the propulsion checkout and monitoring function due to these demands. The principal information source evaluated for this item was the data bus traffic estimates and computer processing estimates presented in the Phase B system studies. The Phase B Main Engine studies were also examined to identify any significant deviations in data management requirements from the Phase B System studies or the basic study results.

B. TASK 1 RESULTS SUMMARY

1. Discussion - In the review of the Phase B System studies, emphasis was placed on the orbiter. Review of the main engine work was limited to the Rocketdyne main engine study due to the unavailability of the Aerojet and Pratt & Whitney reports. The time originally alloted for the two unavailable studies was primarily utilized in two special interest areas; leakage detection and bearing incipient failure detection, identified in our basic study. The Phase I portion of Contract NAS10-7291 (Pearce & Associates) was previously reviewed; however, that material was re-examined and the subsequent Phase II material was evaluated. In all cases, the review and evaluation of the related studies was limited to those items that impact onboard checkout and monitoring.

2. Conclusions - Review of related Space Shuttle studies reaffirmed the fundamental findings of our basic contract which concluded that "the checkout and monitoring approach is technically feasible, will improve system reliability, simplify ground operations, and reduce turnaround time. The implementation of the function must be controlled by a requirements standard, to ensure that the necessary approach and methodology are utilized, and so that the required degree of propulsion and electronics systems integration is accomplished. The basic design of the propulsion systems must incorporate the checkout and monitoring functional requirements including sensors."

   On an individual basis, each study contributed to our objective of updating the baseline. The documentation reviewed aided us in establishing a better functional description which is reflected in the "Guidelines for Incorporation of the Onboard Checkout and Monitoring Function for the Space Shuttle" - Task II. A summary of the study review and evaluation results is presented in Table II-1.

   Our update of the new technology area is also significant. The evaluation indicated that technology exists for bearing incipient failure detection, and can be applied by analyzing the Space Shuttle's rotating machinery to define the bearings' operating characteristics and incipient failure detection requirements, and then developing the sensor and data handling hardware and techniques commensurate with the requirements.

Our evaluation of leak detection technology and applications resulted in the following conclusions:

- Conventional ground leak checks do not adequately ensure that acceptable leakage rates can be maintained during flight operations. This approach is costly and time-consuming, and therefore, is not compatible with Space Shuttle operational concepts.

- Inflight leak detection devices that can measure component internal leakage have not been developed. Leak detection methods that could locate and measure external leakage inflight have been identified (Table II-12), but have not been developed.

- Inflight hazardous gas detection has been identified as a potential requirement for the orbiter, but equipment is not necessarily available that has the required response time for emergency detection, and that is operable from sea level pressure to vacuum conditions.

- We recommend that inflight leak detection development efforts be pursued, with the objective of producing an integrated approach for accomplishing the functions of internal leakage, external leakage, and inflight hazardous gas detection.

TABLE II-1

TASK 1 REVIEW SUMMARY

REVIEW AND EVALUATION	IMPACT ITEM	RECOMMENDATION
Phase B System Studies NAS8-26016 (MDC) NAS9-10960 (NAR)	MDC - Fault isolation to LRU level would be provided in the maintenance area, only. NAR - Inflight fault isolation would be at least to the functional path level. Provisions would be made for fault isolation to LRU level on the ground.	Retain Basic Study concept that fault isolation to the LRU level can and should be done inflight.
	Main engine data used solely for postflight evaluation should be shipped to maintenance recorder via independent transmission link.	Update baseline to incorporate this concept. This concept reduces data bus loading, allowing greater capabilities for handling contingencies.
	Control of individual components.	Identify that this capability is included in the baseline. Individual component operation is required for redundancy verification and maintenance retest.
Phase B Main Engine Study NAS8-26187 (Rocketdyne)	Rocketdyne recommended color change tape and elastomer paint as prime methods for inflight leak detection. These items were recommended for technology development along with integrated sensor packaging and bearing condition monitoring.	It is recommended that the color change tapes and elastomer paints be incorporated as leak detection methods on small lines where no other inflight leak detection devices are available. This technique requires postflight inspection of the treated connections. It is also recommended that passive leak detection be added as a technology development item.
Data Bus Techniques NAS8-26378 (SCI)	No impact items were identified in the course of the review. Items of significance were: (1) an improved data presentation method, (2) number of SSME parameters transmitted over data bus, (3) use of two cable channeling arrangement.	Retain fundamental data bus concepts identified by basic study. Analyze areas not covered by basic study for actual bus design, i.e., environment, statistical reliability and redundancy. Incorporate system improvement items.
Cryogenic Fluid System Optimization NAS9-11330 (Lockheed)	The study review did not identify any major impact items. Several new technology areas were identified which included leakage detection and long life pressure and temperature sensors.	Update sensor technology recommendations to include long life as a general technology goal.
Readiness Assessment NAS10-7145 (GE)	The major impact of this study is the confirmation of techniques and approaches for readiness assessment of mechanical components identified by our basic study.	The concepts and approaches are valid as baselined.
Leakage Detection Improvement Methods NAS10-7291 (Pearce)	No major impact.	Existing data is not sufficient to determine the full potential of ultrasonic equipment for Space Shuttle applications. Ultrasonic leak detection offers promise for inflight detection of internal leakage, providing that the necessary leakage signatures can be established.
Propellant & Gas Handling NAS10-7258 (MMC)	Several technology items were identified for zero-g propellant quantity sensing: (1) Resonant Infrasonics (2) Nucleonic Gaging (3) RF Resonator	No change to baseline. Identification of a firm requirement for zero-g propellant quantity gaging should precede initiation of development. Gaging during vehicle acceleration periods appears more practical due to the development costs associated with zero-g system.
Bearing Incipient Failure Detection (Contract NAS8-25619 extension)	Techniques are available for bearing incipient failure detection. The techniques require development to meet the Space Shuttle requirements. Ability to predict remaining bearing life should be a technology goal.	It is recommended that specific requirements for the application be established, and that a development program be pursued.
Leak Detection Technology (Contract NAS8-25619 extension)	Leakage detection, hazardous gas, internal and external when applied to flight and ground operations cannot be handled by a single existing leakage detection system. Passive leakage detection methods (UV spectroscopy, polymerics, color change tapes and ultrasonics) require further development. Thermal conductivity detectors should be developed together with dual seal flanges as a leak detection technique for large lines.	Update basic study to reflect dual seal technique. Recommend development of thermal conductivity detector, polymerics, color change tapes, and UV spectroscopy for use in detecting external leakage and for hazardous gas monitoring. Continue efforts in developing ultrasonics for internal leakage detection.

C. REVIEWS OF SYSTEMS STUDIES

1. Contracts NAS8-26016 (MDC) and NAS9-10960 (NAR)

Title: Space Shuttle Phase B System Definition Studies.

Study Objective: To analyze and provide a preliminary design for a completely reusable two stage Space Shuttle.

Scope of Review: With the addition of Task 3 to the original plan for the extension to Contract NAS8-25619, it was decided that the review of the Phase B System Studies should primarily encompass the orbiter, since Task 3 was addressed to an expendable booster. Therefore, this review has encompassed the following documents:

NAS8-26016 (MDC)

Document Number	Title
MDC E0308 Part I	Executive Summary
MDC E0308 Part II-1	Technical Summary - Shuttle System
MDC E0308 Part II-2(A)	Technical Summary - Orbiter
MDC E0308 Part II-2(B)	Technical Summary - Orbiter
MDC E0395 Supplement 1	Orbiter Data List

NAS9-10960 (NAR)

Document Number	Title
SD 71-114-1	Volume I, Executive Summary
SD 71-114-2	Volume II, Technical Summary; Book 1, Space Shuttle Program Definition
SD 71-114-2	Volume II, Technical Summary; Book 2, Orbiter Vehicle Definition (Part 1 of 2)
SD 71-114-2	Volume II, Technical Summary; Book 2, Orbiter Vehicle Definition (Part 2 of 2)

The review of this documentation has been confined to the orbiter propulsion systems, the propulsion system instrumentation, and those onboard and ground systems related to the onboard checkout and monitoring functions of the propulsion systems. The review of the propulsion system requirements and design was to assess the impact of differences between them and the NAS8-25619 baseline on the onboard checkout and monitoring function requirements that were identified in our basic contract.

<u>Results of Review</u>: The results of the review of the Phase B System Studies are summarized and compared to the results of basic Contract NAS8-25619 in Table II-2. Significant differences and/or recommendations are noted where applicable. In summary:

- No propulsion system differences were found that had s significant impact on the OCMF requirements.

- It is recommended the NAS8-25619 baseline be updated to specify that main engine data used solely for postflight evaluation be shipped to the maintenance recorder via an independent transmission link to allow greater data bus capability to handle contingencies.

- Two items of significance were found in OCMF concepts. First, our basic study indicated that fault isolation to the LRU level can and should be done inflight. The MDC study indicated that LRU fault isolation capability would only be provided in the maintenance area. NAR indicated that inflight fault isolation capability would be at least to the functional path level and that capability would be provided on the ground for LRU fault isolation but the decision on whether or not the software would be ground or airborne would be deferred. For the reasons noted in Table II-2 (page II-14), our baseline emphasis of inflight fault isolation to the LRU level is valid.

- Secondly, the capability for discreet component control for special tests, such as valve sequencing, was not specifically identified in the basic study. This capability should be reflected in the updated baseline.

TABLE II-2

SUMMARY OF PHASE B SYSTEMS STUDIES REVIEW

ITEM	NAS8-25619 (MMC)	NAS8-26016 (MDC)	NAS9-10960 (NAR)	SIGNIFICANCE/RECOMMENDATION
1. OBJECTIVE	To develop analytical techniques and apply them in defining an approach for accomplishing the checkout and monitoring functions of the Space Shuttle propulsion systems.	To analyze and provide a preliminary design for a completely reuseable two stage Space Shuttle.	To analyze and provide a preliminary design for a completely reuseable two stage Space Shuttle.	
2. MAIN ENGINES a) Number b) Type c) Thrust d) Expansion Ratio e) Propellants f) Mixture Ratio g) Emergency Power Level h) Minimum Power Level i) Inlet Conditions j) Autogenous Output k) Specific Impulse	2 Pump-fed, high pressure, staged combustion. 462 K lb. (vacuum) 400 K lb. (sea level) 154:1 LO_2/LH_2 6.0:1 115% 50% LOX: 164°R; 100 psia LH_2: 37°R; 30 psia GO_2: 4.4 lb/sec GH_2: 1.7 lb/sec Not specified	2 Pump fed, high pressure, staged combustion. 632 K lb (vacuum) 550 K lb (sea level) 150:1 LO_2/LH_2 6.0:1 109% 50% LOX: 170.5°R (max.); 164°R, 100 psia (std.) LH_2: 40.2°R (max.); 37°R, 30 psia (std.) GO_2: 2.3 lb/sec GH_2: 0.519 lb/sec 459 sec	2 Pump fed, high pressure, Pc(nom) = 3,000 psi, staged combusion. 632 K lb (vacuum) 550 K lb (sea level) 150:1 LO_2/LH_2 6.0:1 109% 50% LOX: 162-171°R LH_2: 36-42°R GO_2: 6.0 lb/sec GH_2: 1.7 lb/sec 459 sec	No significant impact on the C&MP.
3. PROPELLANT UTILIZATION AND GAGING (Main Propulsion)	Uses point sensors in the main tanks for loading control, thrust termination prediction, and engine shutdown and depletion.	Closed loop PU system that can accommodate EMR and loading variations and provide a three sigma residual that will be less than 0.25% of total load. Uses segmented, stagger mounted capacitance probes for the main tanks. Point sensors on each segment indicate to the computer which segments are covered. EMR is adjusted based on computer computations on the propellant level data. Loading and low level sensors are used to back up the capacitance probe. A depletion sensor is located in each LOX feedline to ensure that the engine does not experience LOX starvation.	Capacitance and point sensor gaging systems provide redundant propellant loading control signals to ground system. A closed loop PU system consisting of capacitance gaging with back up from a point sensor system supply propellant mass data to the onboard computer which determines the appropriate EMR command to ensure simultaneous propellant depletion. Main engine cuttoff normally initiated by a velocity signal from guidance and control. If propellant is depleted before a velocity cutoff is achieved, the engines will be shut down by the Engine Cutoff Systems (ECO). The ECO system is independent of the PU system and consists of clusters of 5 point sensors in the LH_2 tank outlet and 5 in the LO_2 manifold feedline. ECO electronics are remotely located and signal cutoff on a 2 of 5 voting basis.	Closed loop PU with the accuracy specified for the Phase B systems was not a requirement in the NAS8-25619 baseline. The inclusion of such a system would result primarily in an increase in data bus traffic, computer processing, and calibration requirements. This increase would not be significant.
4. PRESTART LIQUID PROPELLANT CONDITIONING (Main Propulsion)	Prestart propellant conditioning is accomplished by helium injection for a short period prior to engine ignition. Ambient helium is satisfactory for the LOX system; precooled helium is required for the LH_2 suction lines. Geyser suppression is also controlled by helium injection.	Propellant circulation for feedline and engine inlet thermal conditioning is accomplished by triply redundant pumps that force circulation through the engine inlets. The LOX pumps are vehicle mounted while the LH_2 pumps are submerged in the main LH_2 tank. Geyser suppression relies on natural circulation which is initiated by helium injection.	Propellants are recirculated through the feedlines, engines and tanks from prelaunch to engine start. LOX recirculation is by natural convection. LH_2 flow is induced by operation of the engine low pressure turbopump with an electric motor auxiliary drive.	The Phase B systems have greater C&MP requirements to the extent imposed by the additional motors and pumps. These differences are not significant.
5. PROPELLANT DUMP PROVISIONS (Main Propulsion)	No propellant dump provisions were made in the baseline design.	Residual propellants are sequentially dumped through the main engines after engine cutoff. Dump is terminated when tank pressures reach 17 psia. Dump rates are 61 lb/sec/engine for LOX and 30.5 lb/sec/engine for LH_2. Resultant maximum dump times are 76 and 20 seconds respectively.	Propellant dumping through the main engines is a crew controlled operation starting 5 seconds after main engine shutdown.	Recommend that through-the-engine sequential dump provisions be added to the NAS8-25619 baseline propulsion system for landing safety and to minimize OMS/ACPS propellant consumption. No significant impact on the checkout and monitoring requirements would result.
6. POGO SUPPRESSION	POGO suppression was not included in the design reference model.	A requirement for POGO suppression on the long LOX feedlines was determined. POGO suppression was implemented by small passive accumulators.	Not specified. To be defined later.	A passive POGO suppression system such as that implemented by NAS8-26016 would add to the checkout and monitoring function to the extent that is required to ensure proper precharge pressure prior to start.

TABLE II-2 (CONTINUED)

SUMMARY OF PHASE B SYSTEMS STUDIES REVIEW

ITEM	NAS8-25619 (MMC)	NAS8-26016 (MDC)	NAS9-10960 (NAR)	SIGNIFICANCE/RECOMMENDATION
7. ACPS ENGINES				
a) Number	33	30	29	No significant impact on OCMF.
b) Type	Fixed nozzle, film cooled	High pressure, partially regen cooled.	Fixed nozzle, GH_2 cooled	
c) Propellants	GO_2/GH_2	GO_2/GH_2	GO_2/GH_2	
d) Thrust	1,000 lb	1600 lb	2100 lb	
e) Chamber Pressure	300 psia	500 psia	Not specified	
f) Expansion Ratio	40:1	40:1	20:1	
g) Mixture Ratio	4.0:1	4.5:1	4.0:1	
h) Inlet Conditions	GH_2: 500°R; 375 psia GO_2: 500°R; 375 psia	GH_2: 200°R min; 700 psia GO_2: 300°R min; 700 psia	GH_2: 250-500°R; 390 ± 10 psia GO_2: 300-500°R; 390 ± 10 psia	
i) Specific Impulse	Not Specified	435 sec (steady-state), 413 sec (pulsing)	425 sec (min.)	
j) Minimum Impulse Bit	50 lb-sec	Not specified	210 lb-sec	
8. OMS ENGINES				
a) Number	4	2	3	The differences in total available thrust are due to differences in vehicle and subsystem design approaches. The checkout and monitoring requirements of the Phase B systems are slightly greater due to increased complexity in the OMS feed system. The impact on the total OCMF is not significant.
b) Type	Fixed nozzle, film cooled	Gimballed, pump-fed, RL-10A-3-3.	Gimballed, pump-fed, regen-cooled TCA, radiation cooled nozzle extension.	
c) Propellants	GO_2/GH_2	LO_2/LH_2	LO_2/LH_2	
d) Thrust	1.5 K lb (vac)	15K lb (vac)	10K lb (vac)	
e) Mixture Ratio	4.0:1	5.0:1	6.0:1	
f) Specific Impulse	435 sec	444 sec	455.4 nominal (vac)	
g) Chamber Pressure	300 psia	Not specified	800 psia	
h) Nozzle Area Ratio	40:1	57:1	255:1	
9. AIRBREATHING ENGINES				
a) Number	3	4 (5 for ferry missions)	4 (5 for ferry missions)	Recommend that the NAS8-25619 baseline be updated to reflect JP fuel for the airbreathing engines. Engine checkout and monitoring would not be impacted by this change, however, the OCMF requirements of the propellant conditioning equipment for JP is less than for LH_2.
b) Type	Twin spool, non-augmented retractable turbofans.	F101/F12B-3, non-augmented, retractable, removable turbofans.	JTF22A-4, non-augmented, retractable, removable, turbofans.	
c) Fuel	LH_2	JP	JP	
d) Airstart	Windmill; cartridge back-up	Windmill; cartridge back-up.	GH_2 powered turbine assist.	
10. DATA MANAGEMENT & CONTROL a) Central Computer Complex	4 Computers: All computers interface with 4 data buses. 3 computers operate synchronously with 1 computer in control. 4 Bus Output Control (BOC) Units: The BOC units provide data bus switching, memory to data bus data transfers, and data error detection for their respective computers. 1 Memory Access Register Computer (MARC): Votes the operation of the computers and manages computer redundancy under automatic or crew control. Computer Memory: It was assumed that each of the 4 computers had a memory size of 64K thirty-two bit words. Estimated 9K 32 bit locations required for propulsion. Mass Memory: Sufficient storage capability was assumed to exist such that it was not a checkout and monitoring function limitation.	4 Computers: All computers interface with 4 data buses. 2 to 4 computers operate in essential synchonism with 1 computer in control. 4 Input/Output Control Units (IOCU): The IOCUs provide data bus switching, memory to data bus data transfers, and data error detection for their respective computers. 1 System Control Unit (SCU): Votes on computer outputs and manages the redundant computers (with crew override). Provides system status to the crew. Computer Memory: 65, 536 thirty-two bit words. Mass Memory: Used for storing mission planning and prelaunch checkout programs. Storage requirement estimated at 1.2 x 10^7 bits.	4 Computers: All computers interface with 5 data buses. 2 computers are active in a master-slave configuration 4 Bus Control Units (BCU): Operates multiplexer-type I/O channel and interfaces and controls the data bus. Computer control not specified. A master timing unit maintains computer sync. Each central processing unit has 12 main storage units each consisting of 8192 thirty-six-bit words. Mass Memory: Contains program modules and retains selected sets of mission data. Consists of a triply redundant rotating drum with provisions for 400,000 words (32 bit + parity + storage protect).	a) The differences in Central Computer Complexes are considered to be insignificant to propulsion system OCMF.
b) Data Bus Configuration and Characteristics	4 twisted shielded pairs. 1 megabit bit rate. Biphase modulation. Central timing, self-clocking.	4 dual twisted shielded pairs. 1 megabit bit rate. Manchester code. Central timing.	5 twisted shielded pairs. 1 megabit bit rate. Biphase-L Manchester code. Central timing, self-clocking.	b) NAR bus has one more level of fault tolerance; others meet requirements.
c) Data Bus Operation	Time-shared, single active cable, command-response, half-duplex operation under central control. Supervisory and message info from the central computer are sent to remote units. Responses follow over the same cable.	Time-division multiplexed, simultaneous 4 bus operation under central control. 2TSP per bus; one carries data and clock in a half-duplex mode, the other carries timing info in a simplex mode.	Time-division, two active bus, simplex operation under central control. One cable carries clock and command info while the other carries response data.	c) NAR and MMC are more flexible than MDC. MDC has simpler software. Of the three, NAR is preferred.

TABLE II-2 (CONTINUED)

SUMMARY OF PHASE B SYSTEMS STUDIES REVIEW

ITEM	NAS8-25619 (MMC)	NAS8-26016 (MDC)	NAS9-10960 (NAR)	SIGNIFICANCE/RECOMMENDATION
d) Remote Units	Digital Interface Units (DIUs) designed on a minimum hardware philosophy. 25DIUs of 5 basic types are dedicated to the propulsion subsystems to service 322 propulsion LRUs. DIUs contain no processing capability. Each DIU interfaces with all 4 data buses. The propulsion dedicated DIUs average 55 inputs.	64 Digital Interface Units (DIUs) of 4 types service the orbiter's 1100 LRUs. DIUs employ modular construction. Each DIU interfaces with only one data bus to avoid common failure modes.	150 Acquisition, Control and Test (ACT) units (expandable to 254 units) service the orbiter subsystems. Each ACT interfaces all 5 data buses via line coupling units. Each ACT can accept 4 parallel digital words, 1 serial digital input, and 28 analog inputs from the subsystems. Signal conditioning and special conversions are done external to the ACT.	d) The NAR and MMC units are more tolerant of a data bus failure and are more closely dedicated to subsystems or functional paths on a 1-1 basis than MDC units. Therefore, in general, less subsystem capability is lost for a data bus or remote unit failure in the NAR and MMC systems.
e) Processing Time Requirements	The peak computer processing load due to the propulsion system was estimated to be 115 msec/sec. (Based on 2 sec add time machine). This peak occurs during main engine operation.	The peak computer excution time for the orbiter vehicle was calculated to be 461.3 msec/sec. This peak occurs during ascent.	Not specified.	e) Not significant to propulsion OCMF.
11. MAINTENANCE RECORDER	A maintenance recorder was baselined to record trend, performance, and malfunction data for the propulsion system. Recording capacity was assumed to exist to accommodate a recording rate of 15KBPS during a 170 second peak period. Rates at other times are minimal.	2 maintenance recorders are each connected to 2 data buses via 2 DIUs. They store the entire data bus traffic during boost, rendezvous, reentry, and final landing. Trend data from the main engines are recorded redundantly. Stored data are played back after mission completion for trend analysis and maintenance. The maintenance recording requirement is 7.2×10^9 bits.	Main engine data is shipped to the maintenance recorder via hardwire. Snapshot data bus data is inputted via an ACT Select Buffer. All main engine data is recorded during the 270 second launch period. Snapshot data is recorded throughout the mission for trend analysis and maintenance of LRUs. Subsystem malfunction and status data are recorded in CPU main memory.	This item is of potential significance to the propulsion OCMF. The NAR system ships SSME data to the maintenance recorder via hardwire thereby alleviating the data bus of a substantial volume of traffic. The NAR technique is preferrable because it would allow greater data bus flexibility to handle contingencies.
12. DATA BUS TRAFFIC AND DATA SOURCES	The total number of propulsion signals was determined to be 1348. The peak data bus traffic from the orbiter propulsion system occurs during main engine burn and has a value of 64.1 KBPS. This total was based on nominal system operation.	The total number of signals from all vehicle subsystems is 10,398. The total number of propulsion system signals (excluding those assigned to flight control) is 1490. The maximum data rate per bus (from all subsystems) is 155.9 KBPS which occurs during prelaunch. The highest data rate for all 4 data buses is 547.7 KBPS during prelaunch.	The total number of measurements from all orbiter subsystems is 4593. The total number of propulsion measurements is 1387. Resultant data bus traffic has not been specified.	The numbers of propulsion system measurements are comparable even though MMC included inflight LRU fault isolation while the Phase B studies did not. Data bus traffic estimate differences are not considered significant to propulsion OCMF.
13. ONBOARD CHECKOUT AND MONITORING FUNCTION CONCEPTS a) Preflight Checkout	-Preflight checkout of mechanical elements of the propulsion systems will be limited to verification of correct initial conditions for start, and to monitoring of the start-up and operation of those subsystems which are started prior to flight. This criteria is based on the premise that considerable confidence that no failures exist or will occur in the next flight will have been obtained by factory acceptance testing, comprehensive inflight monitoring, postflight evaluation of flight performance, post-maintenance retest of replaced LRUs, and monitoring during ground servicing operations including leakage checks. If the mechanical equipment performed correctly throughout all its previous normal operating cycles (without evidence of failure, degradation, over-stress, or approach of end of normal lifetime) then no significantly greater confidence that it will again perform correctly can be obtained by testing.	-During the prelaunch and launch timeframe the onboard system shall have capability to: control the onboard systems used in fuel loading; provide control and display for subsystem checkout and redundancy verification; control the onboard computer from remote locations; automatically evaluate subsystem parameters; control individual subsystem tests; provide a simulated mission test sequence; verify subsystem performance; provide LRU fault isolation/dignostic routines; monitor subsystems for emergency caution and warning functions. The onboard checkout capability is expanded beyond the flight requirements by using a different set of software. The onboard capability for data requisition, evaluation and display is expanded to permit fault isolation to the LRU, for maintenance purposes, and to perform automatic prelaunch/launch checkout. In general, normal checkout requirements are satisfied by the onboard system with minimal GSE. The propulsion system will use GSE for pneumatics, power, hydraulic power, and leak detection. Provisions for discrete component control will be provided for special tests such as dry leak and valve sequences. Onboard checkout for prelaunch evaluation of the subsystem shall be made only to the level necessary to derive a go/no-go.	-The orbiter shall be capable of self-checkout with onboard systems. For ground turnaround phases, subsystem-oriented checkout programs are provided for LRU replacement verification or to investigate anomalies detected but not resolved in flight. Data acquisition hardware should be onboard, including that equipment required solely for ground checkout.	-It is recommended that capability for discrete component control be added to the NAS8-25619 baseline for special testing such as valve operation for redundancy verification and maintenance retest.

TABLE II-2 (CONTINUED)

SUMMARY OF PHASE B SYSTEMS STUDIES REVIEW

ITEM	NAS8-25619 (MMC)	NAS8-26016 (MDC)	NAS9-10960 (NAR)	SIGNIFICANCE/RECOMMENDATION
	-Preflight self-checks of electronic subsystems and elements will be performed, including verification of sensor electrical elements. This is necessary to achieve a high level of confidence that the electronic equipment is in a flight ready condition, i.e., confidence that inflight failures and indications of impending failures will be detected and isolated.	-Avionics subsystems are checked out with built-in test equipment, built-in test, self-test, monitoring, voting, automatic calibration, and crew evaluation. Prelaunch checkout of the controls and displays subsystem, for example, is an abbreviated version of the ground test/maintenance checkout. Most electronic equipment has BITE for stimuli generation and require no support GSE.	-The verification of proper DCM operation is by a self-test and warning function. Remote units have self-check capability. C&D has self-test for wire-runs, components, and functions critical to flight. Signal conditioning hardware is calibrated prior to the start of a mission and immediately after landing. Propellant gaging systems have provisions for in situ malfunction detection and calibration.	-None
	- Applicable system parameters will be monitored and evaluated by onboard equipment during ground operations for purposes of fault detection, fault isolation, and operating history recording. This operation is augmented by the use of limited GSE such as ground control and display capability during propellant loading.	- During the prelaunch/launch time frame, the onboard system provides control, automatic subsystem parameter evaluation, simulation, fault isolation/diagnostics, and monitoring of subsystems for caution and warning functions.	• The DCM shall provide capability to perform both inflight and ground checkout and fault isolation of all Shuttle subsystems, including itself. The DCM subsystem shall monitor vehicle systems and program orbiter system operations accordingly. The principal vehicle systems are continuously monitored for fault detection and caution and warning.	- None
b) In-Flight Monitoring	Performance and condition indicating parameters will be monitored and evaluated during flight for the following purposes:			
	• In-Flight Ready-To-Start Condition Verification: Appropriate onboard monitoring and evaluation will be provided to verify, just prior to in-flight start, that all applicable equipment and associated system parameters are in the correct conditions for start.	- Not specifically addressed.	- Not specifically addressed.	- None, while the Phase B System studies do not specifically identify this item, it is included in their inflight checkout and monitoring. During this study extension we have lumped this item into a category known as Prestart Checkout that covers both ground and flight phases.
	• Emergency Detection: Provisions shall be made to detect any condition requiring automatic action to avoid a potentially catastrophic effect or requiring special precautions or emergency procedures by the crew. The following criteria apply: . Emergency detection provisions must be redundant. . Redundant caution and warning display capability will be provided for loss or impending loss of major functions, flight safety parameters exceeding safe limits, redundancy reduced to "safe" level, and hazardous leakage.	- During flight the onboard system shall provide display of critical parameters requiring crew action, emergency caution and warning monitoring of all subsystems as required, and subsystem self-test and/or evaluation for time critical subsystem switching.	- Provisions are made to provide both manual and automatic control of subsystem configuration and sequencing of subsystem operations and modes for both normal and contingent operations. Discrete no-go conditions alarm signals alert the flight crew to critical malfunctions. Displays has sufficient redundancy to assure operational condition after any failure. A diagnostic routine for critical failure evaluation will normally be maintained in the main core program. All potentially critical systems or operating conditions shall be monitored and displayed to the crew (and/or computer) when time required for corrective action is critical.	- None
	- Fault Detection: Provisions will be made for inflight detection of failures (including out of specification performance, incipient failures, and transient or intermittent faults) for all identifiable failure modes for which suitable onboard detection techniques and requipment are practical. Fault detection information is required for fault isolation, switching to redundant elements, and for maintenance.	- Failure detection, isolation and switching are performed essentially in real time. Failure detection is accomplished by onboard systems employing a number of means including BIT, BITE, self-check, voting, and performance monitoring and evaluation. The principal uses of inflight fault detection are redundancy management and system reconfiguration, subsystem evaluation, and automatic crew alert for detected malfunctions.	- Vehicle systems shall be monitored for configuration and sequencing control. Faults are detected by performance monitoring of approximately 1,000 significant functional path parameters. Significant propulsion parameters are monitored to detect developing anomolies via trend analysis. Monitoring is normally automatic with reconfiguration generally manually initiated.	• None

TABLE II-2 (CONTINUED)

SUMMARY OF PHASE B SYSTEMS STUDIES REVIEW

ITEM	NAS8-25619 (MMC)	NAS8-26016 (MDC)	NAS9-10960 (NAR)	SIGNIFICANCE/RECOMMENDATION
b. In-Flight Monitoring (Continued)	- Fault Isolation: The criteria established for fault isolation was: . Data for fault isolation will be acquired in flight, by onboard equipment. . Diagnosis for fault isolation will be accomplished with onboard equipment. This diagnosis will isolate faults to the line replaceable unit, and record the data necessary to provide postflight identification of maintenance requirements. . Fault isolation will be accomplished as soon after detection as is necessary to identify lost redundancy and to initiate corrective or safing action when applicable. In cases when no LRU-level redundancy exists and where corrective or safing action is taken in response to failure detection only, as in the case of ACPS engine emergency shutdown, diagnosis for fault isolation may be performed on stored data at a later time. Preferably, this type of diagnosis will be accomplished prior to landing, but may be delayed until after landing if necessary. This criterion was established because in many cases the conditions before, during, and after a failure must be known in order to distinguish between cause and effect, and certain faults may not be isolatable in later tests without resorting to highly realistic simulations of conditions and events.	- Inflight fault isolation is done for redundancy management and system reconfiguration. Detailed evaluation at the subsystem level for fault isolation to the LRU is provided for during the maintenance cycle. The ground fault isolation to an LRU uses a different set of software (modified flight software in some cases). The inflight software is structured to facilitate subsystem monitoring and fault detection. Diagnostic routines are stored in mass storage. The majority of the inflight checkout software is applicable to ground tests.	- Inflight fault isolation shall be accomplished to the functional path as a minimum. (A functional path is the level at which subsystem redundancy is provided.) Fault isolation capability on the ground shall be to the LRU. Access points for LRU fault isolation should be provided but the onboard software implementation decision will be deferred. The onboard COFI program (stored in mass memory) will isolate indicated failures and allow for automatic switching of failed, time-critical, functional paths. The configuration management program will be a subset of the COFI program. A diagnostic routine for critical failure evaluation will normally be maintained in the main core program	- The cost effectiveness of the operational Space Shuttle would be significantly enhanced by performing fault isolation to the LRU level inflight to the maximum possible extent. Generally, it is a matter of generating additional flight software to fault isolate the propulsion system to the LRU level in flight. Neither the magnitude nor complexity of that additional software is prohibitive considering the fact that the bulk of what is needed is already available for performance monitoring, subsystem evaluation, fault detection, emergency detection, caution and warning, redundancy management, reconfiguration management, recording, display and control. We do recognize that certain failures under certain conditions may be more cost effectively isolated on the ground and that it may be desirable to reverify failures in selected LRUs to establish a greater degree of confidence that an element with a long remove, replace, retest cycle is indeed defective. However, inflight fault isolation to the LRU level should be the rule rather than the exception.
	- Real Time Trend Analysis: Real time trend analysis will be performed only for those failure mode cases where it would result in avoidance of significant damage or in early initiation or precautionary action to cope with an impending emergency condition. Real time trend analysis has been limited in this fashion due to the high demands on computer memory size and processing time.	- Not specifically addressed.	- Significant main engine, auxiliary propulsion system, and airbreathing engine parameters will be monitored to allow detection of developing anomalies via trend analysis.	- It was not possible to identify the specific criteria that the Phase B system studies would use for real time trend analysis.
	- Operating Histories: Where correlation exists, or is likely, between an LRU's performance and its operating time, stresses, number of on/off cycles, number of revolutions or strokes, or combinations of these, a history of operation of the LRU will be maintained in computer storage so that when an LRU's operating history exceeds the limit for that LRU, a post-flight printout will provide notification of required replacement. These operating histories will not be continuous, but will be periodic or on-condition updated total of accumulated time, cycles, etc., at discrete states (on, standby, NPL, MPL, etc.) or discrete stress levels (20% overtemperature for example).	- Not specifically addressed.	- Not specifically addressed.	- While the Phase B studies did not specifically address this item, it is likely that this type of data would be extracted from the performance and event data that is recorded.
	- Performance Data: Propulsion systems flight performance data will be acquired and recorded on the vehicle maintenance recorder to enable postflight evaluation for verification of inflight diagnosis, identification of incipient failures, identification and analysis of trends, and to obtain data for design improvement.	- Maintenance data must be recorded during specified mission phases. Critical trend data from the main engines must be redundantly recorded. Data traffic is stored on the maintenance recorder during any equipment malfunction. The stored data is played back after mission completion for maintenance.	- Selected sets of mission data, such as failure data, event data, system performance data, and critical parameter data will be stored in flight. The stored data will provide for postflight analysis of the vehicle performance as an aid in repair and refurbishment.	- None

TABLE 11-2 (Concluded)

SUMMARY OF PHASE B SYSTEMS STUDIES REVIEW

ITEM	NAS8-25619 (MMC)	NAS8-26016 (MDC)	NAS9-10960 (MAR)	SIGNIFICANCE/RECOMMENDATION
c) <u>Postflight Evaluation</u>	-The onboard systems inherently have capability to provide control and monitoring during ground operations such as safing and purging and propellant servicing. They also have processing capability available for use on the ground. This led to the following criteria:			- It is recommended that the NAS8-25619 postflight evaluation criteria be updated to indicate that the onboard processing capability <u>is available</u> for processing of flight recorded data rather than <u>will</u> be used. We continue to recommend the use of onboard capabilities for postflight safing and purging, short term trend analysis, and the identification of maintenance requirements. We recommend that long term trend analysis, such as fleet trends, be done on ground facilities.
	- In-flight type monitoring and evaluation will be continued until completion of shutdown operations. Servicing programs then will be loaded into the onboard computers, replacing the flight programs. Flight-recorded data will be sorted, edited, and evaluated by the onboard computer complex to produce maintenance printouts, trend analysis results and performance data records.	- Data stored on the maintenance recorder is played back after mission completion for maintenance and trend analysis. The onboard system is required to provide capability for safing and purging.	- Postflight analysis will be done on flight recorded data to aid in repair and refurbishment. After a mission completion, the tape can be removed from the airborne maintenance recorder and replayed using standard ground replay facilities. Ground data processing equipment shall be provided as part of the maintenance and repair facility.	
	- Ground connections to the vehicle data bus will be utilized to provide for transmitting commands and data between ground remote control and display provisions and the vehicle central computer complex.	- For ground operations, the data bus has a GSE interface to permit ground monitoring and control as necessary, e.g., during fuel loading.	- For ground operations, the data bus interfaces with the ground system through an ACT/Select Buffer.	- None
d) <u>Maintenance Retest</u>	-Maintenance retest of replaced LRUs will employ the onboard checkout function. The use of the onboard equipment capabilities will minimize GSE requirements. Special ground software is used for this function.	- During the maintenance time frame the onboard system shall provide detailed evaluation at the subsystem level for fault isolation to the LRU, automatic test sequences for special checks such as leak checks, dry gas tests of thrusters, etc., GSE interface to data bus for loading of checkout software for test and control, and parameter monitoring for detailed subsystem checkout and fault isolation. A higher order test oriented language should be used for ground test software. Servicing, maintenance and testing shall not require disturbing the integrity of equipments.	- The orbiter shall be capable of self-checkout with onboard systems. Data acquisition hardware should be onboard, including equipment required solely for ground checkout. Subsystem-oriented checkout programs are provided for LRU replacement verification.	- Recommend that the amount of fault isolation to the LRU required during maintenance be minimized. Recommend adding a higher level test oriented language for ground test use to the NAS8-25619 baseline.
e) <u>Control and Checkout Processing Integration</u>	- Control and checkout will be treated as a single function for purposes of computer processing	- Same as NAS8-25619.	- Same as NAS8-25619.	- None

2. <u>Contract NAS8-26187 (Rocketdyne)</u>

<u>Title</u>: Space Shuttle Main Engine (SSME) Phase B Final Report.

<u>Objective</u>: The major objectives of the Rocketdyne study were to define the SSME requirements, establish engine design, and demonstrate the design feasibility. Supporting analyses and philosophies pertinent to the propulsion and avionics systems are included in the study.

<u>Approach</u>: Analyses and trade studies were performed to optimize engine design. A demonstration firing of the combustion system was conducted to establish the feasibility of the design approach.

<u>Discussion</u>: The final report for the study was issued 23 June 1971. Similar studies were conducted by Pratt and Whitney Aircraft, NAS8-26186 and the Aerojet Liquid Rocket Company, NAS8-26188. The final reports for these two studies were not available for review.

Several significant items applicable to the update of Contract NAS8-25619 were identified in the course of the review. These items as well as applicable recommendations are presented in summary form in Table II-1.

<u>Significance</u>: The most significant area identified by the review is the subject of leak detection. It was found that satisfactory leakage detection techniques are not available for Space Shuttle application. Areas of concern are inflight hazardous gas detection, external system leakage and internal valve leakage. It is recommended that these areas be given study priority.

TABLE II-3

NAS8-26187 REVIEW SUMMARY

PERTINENT COMPARISON ITEMS	ROCKETDYNE, NAS8-26187 SSME PHASE B STUDY	MMC, NAS8-25619 BASIC OCMS STUDY	SIGNIFICANCE	RECOMMENDATIONS
Study Objective	Define and establish SSME design requirements; demonstrate design feasibility and prepare plans for engine development.	Development of analytical techniques for accomplishment of checkout and monitoring functions for the Space Shuttle propulsion systems.	Although the studies are directed toward different goals, there are sufficient similarities in the engine area to make related comparisons.	
Leak Detection				
Inflight	Color change tape and elastomeric paint at joints. Ultrasonic contact probes for engine valve leakage.	Ultrasonic contact probes, ultrasonic microphones and compartment mass spectrometers.	Leakage detection stands as a formidable problem both inflight and during ground operations. The large number of engines, operating conditions, and Space Shuttle requirements disqualify classical leak check operations as the prime method for leakage detection. Suitable alternate techniques are not presently available for Space Shuttle application.	Inflight leakage should be monitored by active monitors which can detect unsafe conditions. Inflight leak detection development should be pursued.
Ground (Scheduled)	100% inspection of joints and evaluation of engine instrumentation.	Analysis of flight data (some helium leak checks required).		
Ground (Unscheduled)	Helium leak detectors, pressure decay; possible ultrasonic contact probes for valve leakage determination.	Helium leak detectors, ultrasonic devices, pressure decay.		
Sensors				
Number and Configuration	104 measurements, dual pressure-temperature sensors utilized. Separated electronics.	150 measurements, dedicated sensors utilized. Recommended investigation of dual sensors.		
Recommended Specialty Sensors	Turbine flowmeter with overspeed brake. Accelerometers for sensing turbopump shaft motion.	Valve position proximity sensors.		
Technology Development Items	Flush mounted LH_2 pressure sensor accuracy and reliability improvement. Passive leak detector, integrated sensor packaging, dynamic strain measuring device for turbopump tests.	SSME fuel flow measuring device (ΔP). Pump bearing condition monitor (acoustic, ultrasonic or deflection method), inflight leak detector (ultrasonics), spark ignition sensor.	A number of technology items are identified which, if fully developed, would benefit the Space Shuttle program.	It is recommended that the following items should be given consideration in the order listed below: Active Leak Detection Passive Leak Detection Bearing Condition Monitors Extended Sensor Operating Life Integrated Sensors
Engine Controller				
Data Sampling Capability	50 samples per second with 50 samples per second transfer rate.	100 samples per second with 50 samples per second transfer rate.	The engine controllers, as described, meet the requirements for the onboard checkout and monitoring functions.	
Features	Dual circuitry (triple input), self-test capability, digital computer, closed loop control, reprogram capabilities, engine checkout and fault isolation provisions.	Dual circuitry, self-test capability, digital computer, closed loop control, reprogram capabilities, engine checkout and fault isolation provisions.		
Internal Inspection	Use of guide tubes and transducer ports for fiber optic boroscope inspection, eddy current internal crack devices, radiation sources for internal X-ray.	Not treated.	Internal inspection provisions supplement the onboard checkout and monitoring function for the SSME.	The internal inspection approach should be adopted in updating the findings of the basic study NAS8-25619.

D. REVIEWS OF TECHNOLOGY STUDIES

1. Contract NAS8-26378 (SCI Electronics, Inc.)

Title: A Study of Multiplex Data Bus Techniques for the Space Shuttle.

Study Objective: The specification of the optimum data bus system for the Space Shuttle.

Scope of Review: This review has been limited to the Phase I Report dated May 17, 1971 and Volume I of the Phase II Report dated January 14, 1972. Volumes II, III, and IV of the Phase II Report, the Phase III Report, and the Final Report were not available at the time of this review.

Study Approach: The approach outlined for this study is to specify a selected data bus system from a number of candidate designs. The selection is to be based on a series of individual studies in which various concepts are evaluated against the Space Shuttle requirements. Six individual studies have been identified from which the design alternatives will be selected. The six studies are summarized in the following paragraphs.

a. Requirements Identification and Analysis - The results of the Phase B system definition studies and NASA inputs are the major sources of requirements for the multiplexed data bus. Those requirements have been recorded on "Space Shuttle Data and Control Requirements Work Sheets" for use in subsequent requirements analysis. Computer punch cards have been made from the work sheets. The punch cards will be used as input data for computer routines from which a variety of listings can be made. Computer routines will be used to process and catalog input data to generate listings of data by: subsystem designation; vehicle location station; message classification; mission phase; signal redundancy level; control and display requirements; and occurrence statistics of aperiodic data. The first listing to be generated accumulated the digital bandwidth required for each subsystem. The ultimate use of the listings will be to generate data flow models that describe the details of each message path between the avionics elements of the Space Shuttle vehicle. The data flow models in turn provide criteria on which candidate system configuration trade-offs can be made. The data flow models will show only normalized raw data flow; overhead cannot be included until a system configuration is selected.

The Phase B vehicle configurations and NASA inputs will also be used to generate physical and environmental models. The principal uses of the physical model will be to determine the cable lengths between locations and the number of data terminals required. The environmental model will identify the temperature, vibration, and EMI levels that the data bus is expected to encounter.

b. Transmission Media - During Phase I of the data bus study, a review of literature related to transmission media was conducted from which inferences were drawn on the suitability of particular media to the Space Shuttle data bus application. That exercise resulted in the reduction of the number of transmission media candidates to two. Wire cable was identified as the prime candidate for the main data bus, while optical links may be considered for special purpose data links. During Phase II, emphasis was placed on laboratory investigations and tests. Tests were conducted to determine characteristics of a number of wire cable candidates, to study operational modes, to select design techniques, to assess EMI impact, and to assess other environmental factors. The final selection was not identified for the transmission media; however, emphasis on wire cable is continuing. Final solutions to the operating mode questions have not been reached; however, bounds were identified for most questions. Conclusions reached in the selection of design techniques were:

   - Transformers are the recommended coupling technique.

   - Cable nonuniformities, or those caused by connectors, are not a significant factor in the transmission of data over the Shuttle data bus.

   - It does not appear that filtering, equalization, predistortion, or inductive loading are necessary to achieve the desired signal characteristics in the Shuttle data bus.

   The most probable sources of EMI to the data bus were identified, the waveforms of impulsive sources were characterized, cable arrangements were evaluated, and the probable effects of EMI on the data bus were evaluated during Phase II.

   It was concluded from the results of these evaluations that:

   - Low pass filtering at the receiver may be an effective method of reducing the magnitude of transient interference.

- Neither radiated interference nor susceptibility present a problem to the operation of a data bus system of the nature envisaged for the Space Shuttle vehicle.

- It does not appear necessary to filter the transmitted data bus signals as a means of reducing radiated interference levels.

Environmental tests led to the conclusions that:

- Added loss incurred at high temperature must be accounted for in the data bus design.

- Changes in characteristic impedance and phase shift with temperature may be ignored when a fluorocarbon insulation is used.

The final selection of the transmission medium will be made from the media candidates and the implementation methods considering length, number of data terminals, bit rate, environment, cost, reliability, availability, etc.

c. <u>Signal Design and Detection</u> - Preliminary signal design and detection studies during Phase I led to the conclusion that time division multiplexing should be the primary multiplexing technique used in the data bus. It was further concluded that digital transmission is the preferred mode of bus operation, that audio and video signals should not be handled by the data bus, and that baseband signaling methods should be emphasized over carrier schemes. The Phase I efforts in this area also led to the following results:

  - The number of baseband modulation techniques for further study was reduced to seven.

  - Emphasis will be placed on block coding over convolutional coding in subsequent evaluations of redundant coding for error control. Elaborate coding for error control does not appear suitable for use on the data bus.

d. <u>Synchronization, Timing and Control</u> - Six major functional categories are identified that comprise the system operation of synchronization, timing and control, namely: timing and synchronization; bus access control; message routing; function and message identification; programming; and channeling arrangements.

A tradeoff between centralized and distributed timing has been identified.

Bus access control using time reference, command-response, handover, and contention techniques are discussed. It has been decided to emphasize the command-response technique during the remainder of the study due to the data flow requirements of the Shuttle and the design approach under consideration.

Five alternatives are presented for the control of message routing under the classifications of centralized and distributed control. The final selection will be made when more definitive data flow requirements have been reached; i.e., the determination of how much of the data must be routed to destinations other than the central processor and how many data transfers require multiple destinations.

Time-reference, function tags, and hybrid identification were considered as possible methods of function and message identification. The latter two are more applicable to the Shuttle and will be emphasized in the later study phases.

A number of applicable alternatives were identified to meet the programming requirements. Overall vehicle software complexity, overall vehicle reliability, and data acquisition and distribution flexibility are key factors that will be considered during subsequent studies of the programming alternatives.

Three 2 cable and two single cable channeling arrangements have been identified for distributing supervisory, synchronization, message, and error control information over the data bus system. The merits of these arrangements will be assessed in Phase II.

e. User Subsystem Interfaces - Studies were conducted during Phase I to identify the functions performed by the user subsystem interface and to identify candidate circuit elements to perform those functions. The electronic operations that must be performed (collection, conversion, formatting, decoding, etc.) and the transfer modes that exist between the data bus subsystem and the user subsystem depend primarily on the subsystem I/O requirements and whether or not the user subsystems possess some or all of the necessary functional capability (such as in an engine controller). A number of

implementation candidates have been identified and will be evaluated pending the definition of the subsystem I/O characteristics from the data flow listings. The identification of several data terminal configurations is planned. They will assume differing channel capacities and mixes of data types. Standardization of data terminals and packaging will be given consideration in subsequent studies.

f. Operational Reliability - Phase I studies of operational reliability addressed:

1) Reliability Criteria

- Failure tolerance criteria
- Statistical reliability criteria

2) Application of Redundancy

- Standby redundancy
- Masking redundancy

3) Design Alternatives for Operational Reliability

- Protection against interference
- Staggering
- Voting
- Channel switching with parity test
- Redundant coding

4) Failure Detection

5) Interfacing of Redundant Subsystems

6) Redundant Clock Sources

7) Power Distribution

8) Status Monitoring and Test

- Requirements
- Built-in test concepts

9) Maintainability

- Built-in test
- Test points
- Modular construction
- Mechanical
- Environmental
- Standardization

It was concluded that a number of functions may be required of the data bus system.

- Staggering and destaggering for protection against interference.
- Automatic error correction and failure masking.
- Fault detection.
- Fault reporting.
- Reconfiguration based on failure detection.
- Interfacing with subsystems having different levels of redundancy.

The evaluation of the effectiveness and complexity of the various techniques and the synthesis of candidate networks capable of satisfying the functional operational reliability requirements have been identified among the Phase II tasks.

<u>Significance to The Propulsion System Checkout and Monitoring Function</u>: In general, the specific implementation of the data bus subsystem will have little impact on the propulsion system checkout and monitoring function assuming that all of the propulsion OCMF requirements have been accommodated. The principal areas of concern are the accommodation of the expected data rate requirements, the action and reaction times required for propulsion system operation and crew safety, the reliability of the data bus to contribute its part to the execution of critical propulsion functions and to preclude the erroneous execution of propulsion functions, and the flexibility to accommodate changes in requirements.

Table II-4 summarizes the areas of mutual interest between basic NAS8-25619 and NAS8-26378. It should be recognized that the differences in scope and objectives of the two studies are diverse. The data bus subsystem proposed by basic NAS8-25619 was part of a design reference model data management system that was a feasible candidate to implement the propulsion system OCMF requirements that had been identified by approaches developed previously in the study. The data bus subsystem that will be specified by NAS8-26378 is intended to be an optimum configuration based on the requirements of all vehicle systems as defined by NASA and the results of the Phase B definition studies. Therefore, while Table II-4 lists a number of areas of mutual interest, our main focus is in the area of data bus requirements due to the propulsion system OCMF (includes control). It should also be noted that the Data and Control Requirements Listing as presented in Appendix B of the Phase I report of NAS8-26378 is the first listing of these requirements and is believed to be preliminary in nature.

The areas related to the data bus in which improvements are recommended to update basic Contract NAS8-25619 are: (See Table II-4)

a. An improved data presentation method;

b. A reconsideration of the number of SSME parameters that should be transmitted over the data bus;

c. The use of a two cable channeling arrangement.

In addition, significant areas in which detailed analyses were not done by basic Contract NAS8-25619 but must be fully analyzed in a data bus design are:

a. Environment;

b. Statistical reliability;

c. Redundancy.

TABLE II-4

SUMMARY OF NAS8-26378 REVIEW

ITEM	NAS8-25619 (MMC)	NAS8-26378 (SCI)	SIGNIFICANCE/RECOMMENDATION
1. OBJECTIVE	Develop and implement approaches for the definition and implementation of the Space Shuttle propulsion system checkout and monitoring functional requirements.	Specify the optimum data bus system for the Space Shuttle vehicle.	
2. SCOPE	Limited to operational phase propulsion system requirements.	Design based on requirements of all vehicle subsystems.	
3. DATA & CONTROL REQUIREMENTS SOURCE	Analysis of baseline propulsion system checkout and monitoring functional requirements for all mission phases.	Phase B Space Shuttle definition study results and NASA inputs.	
4. BASIS FOR SELECTION OF DATA BUS SUBSYSTEM	*Propulsion system requirements and past* experience with onboard checkout systems.	*Evaluation of the results of six indi*vidual studies in which the vehicle data and control requirements are analyzed and candidate designs are evaluated.	*NAS8-26378 data bus design is likely to be better* suited to the entire vehicle since the requirements of all vehicle subsystems will be analyzed.
5. DATA & CONTROL REQUIREMENTS PRESENTATION METHOD	OCMS Measurement Requirements List Subsystem Identity Code Quantity Range and Units Allowable Error Response Rate Mounting Fluid Media Measurement Type Data Use Time of Data Activity Sample Rate Data Rate DIU Number Remarks	Data & Control Requirements Listing Designation Signal Name Classification; Destinations Mission Phase Minimum Samples/Second Location Station Remarks Redundancy Minimum Bits/Word Card Type, Number, and Revision	It is clear that the differences in the presentation methods is primarily a result of the different uses for which the listings were intended. It is suggested that the best of both methods might be merged to create a listing that would serve the purposes of most users. It is recognized that punch card space limitations are a constraint on the amount of information that can be packed into one listing. The recommended listing and rationale follows: *(a) Designation; sufficiently identifies the sub*system, assembly, and signal number, (b) the use of both identity codes and signal names to clearly identify the signal, (c) Range, units, and allowable error; basis for sensor selection, bits/word, and system error budgets, (d) Response rate; basis for sample rate and system speed requirements, (e) Classification; identify signal type, (f) Data use & destination; identification of where the signal is going and for what reason (includes middle column of existing classification column), (g) Mission phase (h) Time of data activity; identify event, condition, interval for which data has significance, (i) Maximum bits/word, (j) Maximum sample rate, (k) Location/terminal, (l) Criticality/redundancy, (m) Remarks, (n) Card mounting and fluid media are more properly listed on sensor criteria or specification sheets. Title and date info should be provided on each sheet of the listing. Onboard Signal List is suggested as a candidate for the title of the listing which should include both measurement (monitor) signals and command (control) signals.
6. a) BOOSTER MAIN ENGINE MEASUREMENTS (not including redundancy)	91/Engine	59/Engine	Rockatdyne indicates that 74 engine parameters result in 109 engine controller inputs, of which 47 are sent over the data bus for recording.
b) BOOSTER MAIN ENGINE RAW MEASUREMENT DATA RATES/ENGINE	9.184 KBPS (Start) 4.448 KBPS (Steady-State) 8.192 KBPS (Shutdown)	11.428 KBPS	The higher NAS8-26387 raw data rates are due to higher transfer rates (not sample rates).
c) BOOSTER MAIN ENGINE TOTAL RAW DATA (includes commands and discretes)	11.640 KBPS (Start) 8.902 KBPS (Steady-State) 10.440 KBPS (Shutdown)		Raw data rates have little significance to data bus sizing. Overhead and command data must be accounted for to determine the peak data rates. NAS8-26378 will do so when a configuration and the operating modes have been selected.
d) BOOSTER MAIN ENGINE DATA RATE PEAK INCLUDING DATA BUS OVERHEAD	30.5 KBPS (Start)		
7. DESTINATIONS OF MAIN ENGINE DATA	Recorder, Displays, Central Computer	Recorder	All SSME data destinations should be accounted for by the data bus study.
8. AIRBREATHING ENGINE MEASUREMENTS	41/Engine	37/Engine	
9. SUBSYSTEM INTERFACE UNITS	25 DIUs (Orbiter Propulsion System) 57 DIUs (Booster Propulsion System)	20 to 100 Data Terminals (DT) per bus.	Depends on interface unit functional capability allocation.
10. DATA BUS			
a) Transmission Medium	*Twisted-shielded wire cable based on* past experience.	*Wire cable for main bus; possibly op*tical link for special purpose links.	
b) Coupling Technique	Transformer	Transformer	
c) Bit Rate	1 megabit bus assumed for a peak propulsion data rate of 430 KBPS.	Considering frequencies as high as 10 MHz.	
d) Environment	Not defined.	Temperature at least 300°F assumed. Bus assumed to be in power switching environment.	Temperature (high & low), vibration, humidity, EMI, etc. must be accounted for in final data bus design.
e) Media Operational Mode	Not specified; have had good success with matched/loaded buses.	TBD	
f) Current/Voltage Operation	Not specified; used voltage in past.	TBD	
g) Multiplexing Technique	Time division	Time division	
h) Transmission Technique	Digital	Digital	
i) Modulation Technique	Baseband (biphase)	Baseband (considering several)	
j) Redundant Coding for Error Control	2 dimensional parity and echo checks.	Emphasizing parity checks and repetition codes.	
k) Message Formats	9 bit bytes; 8 bits data, 1 bit parity. Commands: 1 address byte 1 function code byte 0-7 data bytes 1 parity byte Responses: 1 address byte 0-32 data bytes 1 parity byte	TBD	
l) Timing & Synchronization	Central timing; self-clocking	TBD	

TABLE II-4 (Concluded)
SUMMARY OF NAS8-26378 REVIEW

ITEM	NAS8-25619 (MMC)	NAS8-26378 (SCI)	SIGNIFICANCE/RECOMMENDATION
m) Bus Access Control	Central control on a command-response basis with flexible programming.	Leaning toward command-response approach.	
n) Message Routing	All transmissions occur to or from the central computer complex.	TBD	
o) Function & Message I.D.	Function tags centrally supplied	Function tags or a hybrid technique.	
p) Channeling	Single cable, half-duplex where supervisory info and message info from the central computer are sent to remote units. Responses follow over the same cable.	Considering 5 alternatives.	Recommend that a two cable arrangement be used where supervisory info, sync info, and messages sent from the central computer are sent on one cable with response data sent to the central computer over the second cable. This would double the capacity, simplify the receiver/transmitter circuits and increase the flexibility of the NAS8-25619 baseline system.
q) Failure Tolerance Criteria	FO/FO/FS for transmission media. Variable for remote terminals depending on subsystem redundancy and function criticality. No single point failures.	Considering: No single point failures; FS; FO/FS; and FO/FO/FS as possible criteria for the data bus subsystem elements.	
r) Statistical Reliability	No addressed.	Will be determined.	Must be estimated in addition to failure tolerance criteria to ensure desired reliability.
s) Redundancy	Specified standby redundancy.	Considering standby, masking, or some blend of the two.	The use of standby redundancy should be carefully evaluated in view of the potential time delays in getting the system reconfigured after a failure. Reconfiguration time must be short compared to the reaction time requirements of emergency conditions.
t) Protection Against Interference	Redundant coding; 2 dimensional parity and echo checks of commands.	Considering: staggering; various voter schemes; channel switching with parity test; redundant coding.	
u) Status Monitoring & Test	Centralized monitor and control using parity checks, faulty response, no response, redundant transmission, and self-check methods.	Considering centralized and distributed monitor and control.	
v) Maintainability	Modular construction of 5 basic types of DIUs for the orbiter propulsion system, all with self-check capability.	Should consider: built-in test, test points, modular construction, ease of installation, environmental aspects, and standardization.	The consideration of test points cannot be overemphasized.

2. Contract NAS9-11330 (Lockheed Missiles and Space Co.)

Title: Space Shuttle Cryogenic Supply System Optimization Study.

Objective: Provide sufficient information and recommendations to allow NASA to select the Space Shuttle orbiter cryogenic supply system.

Study Status: At the time of the review, a draft of the interim report had been published and the interim presentation had been given. Work was continuing on the math model.

Approach: The approach taken was to select representative subsystems to supply cryogenics for the main, orbital maneuvering and attitude control propulsion subsystems, and the environmental control, auxiliary power and fuel cell subsystems; and to define purge and inerting subsystems. The next step was to analyze these selected subsystems, define their operating parameters and components, and develop a math model to aid in development of cryogenic subsystems.

Program Summary: The basic requirements and criteria for the cryogenic subsystems were obtained from the main engine and Phase B Shuttle studies. From these studies, propellant quantities, flow rates, pressures, mixture ratios, life, duty cycles, etc., were defined. To formulate the candidate subsystems for evaluation, the subsystems were divided into their main functions which were again divided into increasing detail to provide all possible subsystem compositions. Various options for each function were considered and presented. For each defined function, favorable and unfavorable factors were listed. Candidate systems were formed by picking functions from the various options and eliminating those functions with unfavorable factors from further evaluation.

These candidate systems were then analyzed and when applicable, methods or components were compared, and the favorable and/or unfavorable factors presented. The specific comparisons or analyses are too numerous to list completely, but as an example the following list is presented:

- Compare electrical pumps and turbopumps for the auxiliary power unit.
- Dual or cascaded propellant tanks for the orbital maneuvering subsystem.
- Use of start tanks.

- Effects of insulation thickness.
- Location of pumps.
- Types of insulation.
- Comparative weights.

From these studies, Lockheed supplied Air Research with schematics on candidate subsystems for the various subsystems anticipated for use on the Shuttle orbiter. Received by Air Research were schematics on the following subsystems: environmental control, fuel cell supply, auxiliary power, main propulsion, orbital maneuvering, attitude control and purge system. Also received were duty cycles, size, flow rates and fluid conditions. The major areas of effort for Air Research were: component recommendations, malfunction examination and development of parametric data with major emphasis placed on valves, heat exchanges, pumps and instrumentation.

The component recommendations were related to existing components which would best satisfy Lockheed's input data. Where the recommended component would not meet all the requirements, the necessity for modifications or new technology was noted. In each component identified, a malfunction examination was performed. Criticality of the component was assessed from the effects of its failure on the system and its failure rate. The failure rates used were Apollo failure rates, estimates based on the Apollo failure rates or were calculated from similar equipment. The failure modes and effects were derived from experience since most of the components recommended were made by Air Research. The schematics given were "single string" only, meaning no redundancy was considered. Air Research's component malfunction analysis also made recommendations where redundancy should be considered.

Parametric data was presented on each of the recommended components. This consisted of size, weight, power requirements, pressures, temperature, weight flow, effective flow area, etc.

Heat exchangers and pumps are designed for specific systems, therefore, "off-the-shelf" components could not be recommended. However, to aid in selecting these items, parametric data was derived by varying inlet conditions, flow rates, etc. and showing it in graphic form so that weight, size, efficiency, power requirements, speed, etc., could be determined depending on system requirements.

The primary use for instrumentation considered in Air Research's study was for control purposes, no instrumentation was recommended for monitoring or checkout. No parametric data for instrumentation was presented per Lockheed's directive; however, malfunction data was presented (function, failure rate, failure type, effect and recommendations for redundancy).

<u>OCMF Impact</u>: Onboard checkout and monitoring per se was not considered in the cryogenic supply system optimization study documentation that was reviewed. Instrumentation components (pressure transducers/switches, temperature pickups) considered in this report were for control purposes only; specifically for control of regulating methods, i.e., heaters, fans, pressure regulating, etc.

However, some new technology areas were defined for instrumentation. These areas were identified while recommending components for use in the subsystems being evaluated. Areas in which existing instrumentation did not meet the specific requirements defined in this study are:

a. Pressure switches needed life time improvement for cryogenic applications.

b. Pressure transducers that can function immersed in liquid hydrogen. These are required for monitoring the integrated cryogenic systems without liquid orientation.

c. Leakage detection devices are needed to detect gas losses and safety hazards.

d. Temperature sensors and control logic needs development for controlling venting by sensing liquid temperature.

Although onboard checkout and monitoring requirements were not included in the documentation reviewed, no new requirements for the OCMF were identified by the proposed cryogenic subsystems that were not considered in the basic OCMF study.

3. Contracts NAS10-7145 and -7788 (General Electric)

Title: Study of Techniques for Automatic Self-Contained Readiness Assessment & Fault Isolation for Ground and On-board Mechanical Systems.

Objective: The objective of the basic study (Contract NAS10-7145) was to provide direction for achieving technology by which mechanical devices' performance parameters can automatically, and in real time, be determined and evaluated. The objective of Contract NAS10-7788, which is a follow-on to the basic, is to evaluate the feasibility of structure-borne acoustic techniques for readiness assessment and performance monitoring of mechanical systems.

Status: The basic study has been completed. The effort on the follow-on program was initiated in October, 1971.

APPROACH: A representative sample of mechanical devices were selected for study from Saturn V Launch Complex 39 flight and ground systems. The sample consisted of 36 devices which were subjected to detailed analysis to establish functional parameters necessary for readiness assessments and to determine methods for achieving these capabilities. Each of the 36 devices was analyzed to verify the functional parameters which must be monitored to ascertain whether the device is in a Go, No-Go, or Caution status. The analysis consisted of identifying the devices' failure modes, the basic parameters that could detect each failure mode, and the parameter boundaries which define its status. Seventeen separate requirement parameters were identified for the 36 components. Each of the mechanical devices typically possess from 4 to 10 of these parameters which must be evaluated to achieve total readiness assessment capability. Each of the parameters were compared to non-destructive test techniques. (As part of the study, ninety non-destructive test techniques were identified and categorized according to their adaptability to the mechanical systems readiness assessment function.) Eighteen existing non-destructive test techniques were applicable to the 36 components. Parameters that were not amenable to existing techniques were structural stress, surface smoothness, lubricity, certain distance and force measurements, and leakage. Potentially adaptable techniques recommended for further evaluations were grouped in the following categories:

- Acoustic non-destructive testing
- Infrared Radiometry

- Optical Interferometry
- Ultrasonic Imaging

The effort under the follow-on program is comprised of an evaluation of the applicability of structure borne acoustics. In task I, three different types of mechanical components were selected from the Saturn V spare parts inventory; a solenoid valve, a regulator, and an air motor. These components were analyzed to define failure mechanisms, failure modes, and operational effects. Acoustic signature predictions were derived for each component by the formulation of an acoustical model. Detailed acoustic test plans were then generated. Task II is comprised of conducting acoustic tests on the selected components, evaluating the acoustic test data, and comparing the predicted signatures with the actual signatures. A minimum of three tests will be conducted on each of the components under varying operating conditions. Signatures of induced failure modes will be evaluated against a baseline and the analytical model. The primary output of Task III will be a general Structure Borne Acoustic Methodology Handbook. This handbook is intended as a reference manual for system designers.

Significance of Results: It was concluded from the basic study that existing hardware was not configured for real-time, self-contained status assessment; however, future mechanical devices could be so configured. To implement readiness assessment a balanced program of studies and development programs were recommended to provide the optimum method of achieving this goal. The recommendations generally consisted of software and hardware development to expand the area in which existing technology is not available for readiness assessments.

The structure borne acoustics evaluation that is being conducted under the follow-on program had not progressed to the stage where the applicability of the technique to the propulsion systems onboard checkout and monitoring function could adequately be assessed. However, the technique has been shown to have potential applications for fault detection and for acquisition of trend data.

4. Contract NAS10-7291 (Pearce and Associates; Dynamatec Corp.)

Title: Study to Develop Improved Methods to Detect Leakage In Fluid Systems.

The Dynamatec Corporation study consists of three phases. This review covers the first two phases which have both been completed. Data on the Phase III work has not yet been received. A summation of the first two phases of the study are presented in Table II-4 and II-5.

The objective of Phase I was to develop a faster and more reliable method to detect tank and system leakage. A method was to be developed which would lend itself to onboard checkout and monitoring of Space Shuttle systems. This was to be accomplished by updating present leak detection techniques or by designing a new system. The recommendations presented at completion of Phase I led to the initiation of Phase II.

The objective of Phase II was to design and fabricate a prototype ultrasonic contact sensor leak detector and demonstrate its operation during cryogenic operations at KSC.

The objective of Phase III is to develop flight weight electronics for use with the contact sensors developed in Phase II.

Conclusion: The findings of NAS10-7291 do not alter the basic conclusions of NAS8-25619. Acoustic/ultrasonic techniques appear to hold promise for leakage detection, and should be considered in the development of a detector system for internal and external leak detection. Emphasis should be placed on establishing the requisite leakage signatures.

TABLE II-5

PEARCE PHASE I SUMMARY

ITEMS	REMARKS (TEXT)
SCOPE: Detect leakage in gaseous systems to bubble tight range.	
APPROACH: Conduct industry leak detection survey.	Only Perkin Elmer mass spectrometer seemed to meet any of the requirements established for future applications.
Review leak detection work at Government centers.	Elastomer paint - can complement ultrasonics. Handheld leak detector - does not meet requirements.
Research update data on KSC systems.	Gathered information used to establish future requirements.
Evaluation of future requirements.	Results: 1. Rapid response leak detection system should be capable of detecting leakage in the 10^{-4} cc/sec range. 2. Detection method should sense helium. 3. The leak detection method should lend itself to onboard vehicle checkout.
Evaluation of present systems for future needs.	Present systems do not meet future needs.
Design new system.	Ultrasonics most promising for flight and ground leak detection; mass spectrometer for inflight hazardous gas monitoring.
Automatic compatibility review.	Leak detection system monitored by onboard computer to analyze and determine appropriate action.
Phase I recommendations.	.Use ultrasonics for primary leak detection system. .Use cabin-gas analyzer for onboard hazardous gas detection. .Develop a more sophisticated ultrasonic prototype and demonstrate it under operating conditions. .Expand cabin gas analyzer efforts to achieve compatibility with Space Shuttle requirements. .Nude source ionizer should also be developed for flight usage.

PEARCE PHASE II SUMMARY

ITEMS	REMARKS (TEXT)
SCOPE: Design and fabricate ultrasonic contact sensors. Design supporting electronics (breadboard) and demonstrate packaged prototype.	
DESIGN AND FABRICATION:	
Contact Transducer	Nonfloating piezoelectric crystal for greater sensitivity; Teflon isolated to reduce air transmitted vibrations; Machined aluminum contact surface.
Signal Conditioner Assembly	Five channel, each at specific frequency band (1-100 KHz).
Control and Display Unit	One channel oscilloscope or digital display; audio output capabilities; channel selection 1 thru 5.
LAB TESTS: A series of failed components were tested for leakage. Established leaks were located by water bath. Leaks ranged from 0 to 30 bubbles/sec.	Leaks detected below 10 psi Δ P range; tests demonstrated system flow metering capabilities; N_2 produced more ultrasonic noise than He.
KSC TESTS: Five transducers mounted on H_2 dewar and lines during dewar loading operations. 30 min. of test data recorded.	Detected H_2 phase change in vent line, start of purge, and fill valve opening. Operated at cryogenic temperatures.
SUMMARY:	.Ultrasonics have ability to detect internal leakage, operate in vacuum, monitor valve actuation, measure fluid flow and detect phase change. .Existing state-of-the-art sensors cannot go to bubble tight range. .Detectors should be designed into flight hardware.

5. Contract NAS10-7258 (Martin Marietta Corporation)

Title: Propellants and Gases Handling in Support of Space Shuttle - Cryogenic Propellant Loading; Purging LH_2 Systems; LOX Geyser Suppression; Propellant Tank Level Sensing.

Objective: The objective of this study on Space Shuttle cryogenic propellants was to:

a. Define modifications to the Saturn V launch facility propellant storage and transfer equipment for Space Shuttle.

b. Define techniques for reducing quantities of helium purge gas.

c. Evaluate the feed system design approaches for geysering suppression.

d. Define improved propellant level sensors and quantity gaging systems.

Study Status: The study is complete and the final report was published June 1971. A follow-on study (Contract NAS10-7613) was performed to evaluate geyser suppression techniques. The final report was in draft form at the time of this review.

Approach Of The Review: The major emphasis of the review was placed on the propellant level sensing methods presented, the recommendations made and any effects on onboard checkout concepts.

Report Conclusions; For propellant handling the Saturn V cryogenic loading systems can be modified to provide the increased loading demands of the Space Shuttle vehicle. No difficulties are expected from the modification since the hardware needed is presently available and no new technology is necessary. The costs of the modifications will depend on the loading time selected. The minimum loading was about one hour and costs approximately 370% more than the minimum cost method which took about three hours for propellant loading.

The cost of purging can be greatly reduced by using a two-gas system. It was recommended that GN_2 by used until the systems are prepared for admission of LH_2 at which time GH_2 should be used.

Concentric suction lines were the recommended technique for geyser suppression. This method is self-starting, self-regulating and is not dependent on other active systems, such as pumps. Tests performed under this contract were on straight vertical lines only, and it was recommended that the suitability of this method be demonstrated on more complex line configurations. The previously mentioned follow-on did test this technique on simulated Space Shuttle propellant lines and found it to be a viable technique.

The gaging sensors and systems discussed in this study are summarized in Table II-7. The gaging and sensors recommended for the Space Shuttle are presented in Table II-8. Also recommended was the use of digital solid state electronics for the capacitance probe systems which would reduce the weight from 50 pounds for the Saturn type to about 14 pounds.

Review Observations: No problems are foreseen in integrating the gaging systems with onboard checkout and monitoring. Capacitance probes and point sensors are used on Apollo to provide the crew with propellant quantity readings while the engine (SPS) is operating (propellants are oriented). Problems have been encountered with the capacitance gaging systems both in the booster and the SPS. Contamination and electrical problems have occurred, and the systems have flown inoperable on some missions.

The zero-g methods proposed are not operational and need considerable development. Any potential requirement for a zero-g gaging system should be weighed against the cost, complexity, accuracy, and reliability of such a system before requiring zero-g gaging for the Space Shuttle. Inventory-keeping, needing no technology advancement, may prove to be the optimum system and should definitely be considered. With onboard monitoring and the use of the flight computer an inventory-keeping system should not present any major implementation problems. This technique can be applied to all propellants. The recommended systems and others listed in Table II-8 have limitations as to which propellant (LH_2 or LO_2) that they can be used with, meaning that two systems may have to be developed, further increasing the costs.

TABLE II-7

PROPELLANT GAGING SUMMARY

GAGING SYSTEM OR SENSOR	PRINCIPLE OF OPERATION	PROPELLANT ORIENTATION	EXPERIENCE	ADVANTAGES	DISADVANTAGES
<u>DISCRETE POINT LEVEL SENSORS</u>					
Float Switch	A float with a magnet is used to actuate reed switches inside a stainless steel tube.	Oriented	. THOR . MMC has used a float switch system in liquid fluorine in ground tests.		. Low density of LH_2. . Difficulty of building a sealed float strong enough to withstand high pressures and light enough to float in LH_2. . Requires moving parts in tank.
Capacitance Sensors	The capacitance is measured between two plates in the tank. The difference in dielectric constant between the gas and propellant gives an indication that the plates are in the liquid or gas.	Oriented	S-IC S-IVB	Accurate Can be used in LH_2 & LO_2.	
Thermal Conductivity (Hot Wire Sensor)	A wire whose resistance varies with temperature is used in the tank. Heat dissipation when submerged will change the resistance which is detected by a bridge circuit. Current can be controlled to maintain constant (and safe), temp., with indication achieved by observing current flow.	Oriented	Atlas SII (1) SIVB (1) (1) Engine cut-off and loading (experience has been very good).	Extremely small mass of the sensing element. Can achieve good reaction time (approx. 30 milliseconds). Simple sensing circuit. Accurate (eliminating splashing effects).	
Optical Refraction	Light is directed down a transparent cylindrical prism, if liquid is present light passes out due to similar refraction indices; when liquid is not present light is transmitted back to solar cell giving a "no liquid" indication.	Oriented	S-IC (cut-off system and LO_2 tank overfill indication)	Sensor can be used in both LO_2 & LH_2. Accurate. No need to calibrate for different temp.	
<u>CONTINUOUS GAGING, NOT ZERO-G</u>					
Hydrostatic	The hydrostatic pressure is measured at the bottom of the tank, a ΔP measurement is taken from the tank top when the tanks are pressurized.	Oriented (1g)	Has been used extensively. S-IB; LO_2 & RP-1 loading.	Minimum airborne weight. Continuous gaging. Accurate.	Limited to constant acceleration levels (i.e., loading). Accurate pressure measurement with cryogenics may be difficult especially with LH_2 due to its low density and temp.

TABLE II-7 (Continued)

PROPELLANT GAGING SUMMARY (CONTINUED)

GAGING SYSTEM OR SENSOR	PRINCIPLE OF OPERATION	PROPELLANT ORIENTATION	EXPERIENCE	ADVANTAGES	DISADVANTAGES
Capacitance Probe	Plates are mounted vertically the length of the tank. The dielectric constant difference between liquid and gas changes the capacitance measured between these plates indicating liquid level. The spacing between the probes can be varied with tank geometry to give a direct readout of liquid quantity.	Oriented	Has been used in S-IC, SII and SIVB and Service Propulsion System for propellant utilization. Method has been in use about 25 years.	Continuous gaging.	Potential problem with contamination unless addressed in basic design and by operational contamination control measures.
Sonar	A piezoelectric crystal sends a sound impulse to the liquid surface from the bottom of the tank, the impulse bounces off the liquid-gas interface and is picked up by the transducer and the total time is electronically measured. The speed of sound in the liquid times the time gives the level.	Oriented	Used on Titan at one time.		Inaccurate at low propellant levels.
Continuous Hot Wire	(See Thermal Conductivity write up.) This method is an extension of the point sensor method mentioned before except that a continuous wire is mounted in the tank. The change in resistance is directly related to the amount of wire submersed.	Oriented	Experimental only.	Light and simple.	Thermal expansion of the wire would cause mounting problems. Requires support so propellant motion or vibration will not break wire.
<u>ZERO-G QUANTITY GAGING</u>					
Capacitance Probe	See previous write up.	0-g	Experimental		Requires up to ten probes for $\pm$ 2.5% accuracy, thus weight would become excessive.
Resonant Infrasonic Gaging System (RIGS)	The difference in compressibility between liquid and gases is used for gaging based on the measurement of dynamic pressure generated by periodic volume compression at a suitable frequency. A vibration bellows is used to create a volume variation in the ullage gas. The ullage compressibility changes as a function of total ullage volume.	0-g	Experimental (has been tested on an orbital flight).	Has shown some promise for 0-g gaging.	Acoustic noise and liquid entrapped in the gages presented problems in the flight test. An isolation diaphragm is needed which is a problem for cryogenic applications (no material for LH_2). Problem with different pressurants, i.e., He for LO_2. LH_2 vapor properties do not lend themselves to this technique.

TABLE II-7 (continued)

PROPELLANT GAGING SUMMARY (CONTINUED)

GAGING SYSTEM OR SENSOR	PRINCIPLE OF OPERATION	PROPELLANT ORIENTATION	EXPERIENCE	ADVANTAGES	DISADVANTAGES
Radio Frequency (RF)	Based on the principle that the tank can be made to function as a cavity resonator. The tank is illuminated with resonant frequency, or frequencies, and the resonant modes are monitored.	0-g	Experimental		Tank must be a conducting material suitable for forming a continuous electrical path. Large errors encountered in simulated zero g tests. Tank configurations can be a problem especially ones with screens, baffling, etc. Conductivity of metal is temp. dependent. Cryogenics could be problem.
Nucleonic Gaging	A source and detector are mounted outside the tank and opposite each other. Gamma rays emitted by the source pass through the tank and are counted by the detector. The amount of particles, from a constant source, picked up by the detector is directly proportional to the amount of liquid in the tank.	0-g	Experimental, an operational system has been used for oil quantity gaging.	No moving parts. Nothing placed in the tank (no leak paths). Shows promise as method for measuring LH_2.	Accuracy problems with systems tested, need better detectors and means of calibrating tanks. LO_2 requires very high energy source due to LO_2 absorbsion characteristics (high energy source would be safety hazard).
Trace Gas System	A known amount of trace gas is injected into the ullage and is diffused throughout. As pressurant is added to replace outflowing propellant the concentration decreases in direct proportion to the increased ullage. Concentrations are measured by chemical analysis, radioactivity, etc.	0-g	Experimental	Should work well on bladder and screen tanks.	Cannot be used on vented tank. No tracer gas can be used with LH_2. Must find a "tracer" that will not be absorbed into propellant. Response is currently slow.
Inventory Keeping	Maintain record of propellant loaded and used. Requires use of flow meters or valve traces to keep record of used propellants.	0-g	Concept evaluations have been made, technique has been used in various degrees by all space flights.	Does not require extensive development, technology is available.	Boiling and venting of cryogenics can be problem. Cannot detect leaks.

TABLE II-8

RECOMMENDED GAGING METHODS
FROM MMC STUDY (NAS10-7258)

VEHICLE	REQUIREMENTS (1)		RECOMMENDATIONS
	TANKS	MEASUREMENT FOR:	
Booster	Main LO_2 LH_2	Loading (to ± 1%) Replenishment Depletion	Hot wire sensors. Hot wire sensors. Hot wire sensors.
	Cruise LO_2 RP-1	Loading (to ± 1%) Continuous gaging	Capacitance probes. Capacitance probes.
	ACPS-APU LO_2 & LH_2	Loading (to ± 1%)	Hot wire.
Orbiter	Main LO_2 LH_2	Loading (to ± 1%) Replenishment Depletion Continuous gaging during main engine burn.	Capacitance probes. Capacitance probes. Capacitance probes. Capacitance probes.
	Secondary (ACPS and OMS) LO_2 LH_2	Loading (to ± 1%) Replenishment Continuous or periodic quantity.	Hot wire. Hot wire. RIGS (LO_2), Nucleonic (LH_2)

(1) At the time the study was made.
(Information was received from McDonnell Douglas.)

E. ADDITIONAL INVESTIGATIONS

1. Bearing Incipient Failure Detection: It was reported in the final report (Vol. III, pages III-32, 33) of our basic study that incipient bearing failure detection was one of the identified requirements that was not readily met by the application of existing sensors or developed sensor concepts with onboard equipment. It went on to report that Pratt and Whitney had obtained good results with ultrasonic range accelerometers on turbofan engine bearings and had success with bearing vibration analysis during the RL10 rocket engine development. Other references that were mentioned contained reports on engine vibration monitoring with piezoelectric vibration transducers on the Boeing 747 and finally the acoustic emission studies of H. L. Balderston of the Boeing Company for the detection of incipient failures in structures and various subsystems.

   Since the effects of certain Space Shuttle bearing failures are potentially catastrophic and no known advancement has been made that would preclude bearing failures during a Shuttle mission, we have continued to pursue the subject of incipient bearing failure detection and malfunction detection during this study extension. We have made personal contact with SKF Industries, Inc. and the General Electric Company, both of whom have developed bearing failure detection equipment.

   In our visits with SKF and GE, we provided each with the background of this study, the principal Shuttle applications where incipient bearing failure detection is required, and the relationship of incipient bearing failure detection to onboard checkout and monitoring. In preparation for these visits, we consulted Rocketdyne on the bearing characteristics of their SSME turbopumps since this is an area of particular interest.

   Our visits with GE and SKF have convinced us that each of them has developed valuable techniques that have been sufficiently proven in a wide range of applications to warrant development for the Space Shuttle propulsion system application. Both techniques have been developed from an understanding of the failure mechanisms of rolling bearings and the resulting signatures. Spalling due to metal fatigue on the bearing surfaces result in the generation and transmission of energy impulses (narrow, high amplitude pulses) when a rolling element comes in contact with the damaged area. This fact has led to

the development of equipment to detect these energy impulses while ignoring other inputs. It eliminates the necessity to attempt exotic analyses of complex vibration waveforms and precludes the necessity for extensive testing of acceptable and damaged bearings to identify signatures of damaged bearings. Both techniques use piezoelectric accelerometers for the sensing elements. The accelerometer outputs are fed to signal conditioners where undesired signals are "filtered" out and the health of the bearing is assessed by evaluating the rate and amplitude of the remaining pulses. The differences in the respective techniques is in the signal conditioning equipment. In addition to detecting fatigue damage on the rolling elements, these techniques are capable of detecting lubricant impurities and cage damage. The GE device has also been used to detect out-of-round balls and incorrect loading.

On the basis of the GE and SKF information, it is our conclusion and recommendation that the primary areas requiring development are: (1) the extension of present detection techniques to bearings operating at cryogenic temperatures, (2) the development of these techniques for 2-3 inch bearings operating at varying speeds up to 40,000 rpm, and (3) the development of the capability to predict the minimum remaining useful life from the time that an anomaly is first detected. In conjunction with these efforts, it is recommended that optimum sensor mounting criteria (considering physical environment and detector sensitivity), self-check techniques, sampling rate and signal conditioner sharing criteria, and display and recording criteria be developed for the specific Space Shuttle applications. It is further recommended that appropriate emphasis be placed on failure prevention to limit the need for incipient bearing failure detection capability to only those areas of highest criticality.

In addition, a limited literature search has identified a number of other manufacturers of vibration and vibration/shock measuring equipment. This search was made to identify suppliers of complete units or systems rather than sensors as was the case in the vendor survey in our basic study. Table II-9 lists the other manufacturers identified and the type of equipment that they supply. Most of the equipment of these manufacturers is designed for low speed, non-critical, industrial use and is not likely to have much applicability to the Space Shuttle. However, an exploration of these sources, shown on the table,

C.2

as well as Pratt & Whitney would be advisable in further pursuit of solutions to the bearing incipient failure detection and malfunction detection problems for the Shuttle.

The technical information supplied by Mr. K. Smith of the General Electric Company and Messrs. P. Howard, L. Sibley, and T. Tallian of SKF Industries was a significant contribution to this technology review and is gratefully acknowledged.

TABLE II-9

MANUFACTURERS OF VIBRATION & SHOCK MEASURING EQUIPMENT*

MANUFACTURER	TYPE OF EQUIPMENT	
	VIBRATION	VIB./SHOCK
AGAC - Derritron, Inc. Alexandria, Virginia	X	
Bell & Howell Co. CEC/Transducer Div. Monrovia, California	X	
Boeing Co. Houston, Texas		X
Columbia Research Labs Raytheon Company Sudbury, Massachusettes		X
General Radio Co.	X	
Gulton Industries, Inc. Servonic Instruments Div. Metuchen, New Jersey	X	
Hanchett Mfg. Co. Raydyne Div. Big Rapids, Michigan	X	
Indikon Company, Inc. Watertown, Massachusetts	X	
International Res. & Dev. Corp. Subsid. of H. H. Robertson Co. Worthington, Ohio	X	
Korfund Dynamics Corp. Instruments & Acoustics Div. Westbury, L. I., New York	X	
Mechanical Technology, Inc. Instruments Div. Latham, New York	X	
Reliance Electric Co.	X	
Robertshaw Controls Co. Aeronautical & Instrument Div.		X
Southfield Electronics Div. Comtel Corporation	X	

*In addition to GE and SKF.

2. Leak Detection

The basic study identified a candidate approach for incorporating leak detection into onboard checkout and monitoring of the Space Shuttle propulsion systems. However, the study also identified leak detection as requiring further technology efforts. To supplement the work conducted in the basic study and in the related studies, further work was conducted in surveying and cataloging potential leak detection techniques. The approach and results of this additional activity are presented herein.

Approach: The approach consisted of establishing a leakage criteria baseline to enable evaluating the applicability of various leakage detection techniques; conducting a survey and evaluation of proven and potentially feasible leak detection methods and approaches; and identifying general leakage sources and time phases, and matching to them candidate leakage detection methods.

- Generation of leakage criteria. Leakage criteria data were obtained for a launch vehicle (Titan III) and a spacecraft (Apollo CSM). The information was categorized and, where possible, converted to cubic centimeters per second leakage rates. It was determined that the Space Shuttle Propulsion System leakage would fall within the ranges of the Titan-Apollo criteria. Rough order of magnitude leakage rates were then assigned to component groupings and conditions. The criteria are summarized in Table II-10.

- Survey of leakage detection methods and approaches. A survey of industry, government and scientific sources was conducted to define and determine the status of existing and future leak detection technology. A tabulation of sources of information is presented in Table II-11. Recommendations that update the basic study results are based on the information obtained in the survey.

- Evaluation of leakage sources and detection methods. Leakage sources and conditions were categorized and applied to identified operational time phases. Leakage detection methods were also treated in the same manner. Techniques for detection of specific leakages were recommended on the basis of applicability, availability, and technology status. Where possible a set of three recommendations were made consisting of a preferred state-of-the-art technique, an alternate method, and a technique that potentially would best suit the application but would require a significant development effort. Results of the evaluation are presented in Table II-12.

The following ground rules were used in this investigation:

- Leak checks would not normally be conducted during the vehicle flight preparation operations. However, the leakage detection recommendations defined for maintenance retest operations would be applicable should leak checks be required for normal flight preparations.

- The orbiter was used as a baseline vehicle for leakage detection requirements and approaches.

- Portions of the orbiter propulsion systems are isolated by compartments that are capable of being purged with inert gas during conditions that could constitute a fire hazard.

- The vehicle compartments are vented to space during orbital operations.

- The main propulsion system propellants are liquid oxygen and hydrogen. Hypergolic propellants such as MMH and N_2O_4 could be used by the orbital maneuvering and attitude control systems.

Discussion: The leak detection technology evaluation indicated that there is no single detection method or instrument that fully satisfies the Space Shuttle leakage detection requirements. In fact, a good deal of development work is necessary to have an onboard checkout and monitoring capability sufficient to preclude tedious ground leak check operation. It is our conclusion, however, that with proper design planning, adequate methods can be devised to minimize laborious ground leak checks. To attain this goal, the following recommendations are presented:

a. Propellant and gas lines approximately two inches in diameter and larger should utilize a dual seal flange with a vent cavity monitored by a thermal conductivity leakage sensor. Dual seal was used on Saturn I-C, and was considered for the upper stages of the Saturn vehicle. The dual seal concept with thermal conductivity sensor is presently being evaluated under Contract NAS1-10840 by the McDonnell Douglas Astronautics Company. The application of this approach requires development of a seal concept that will provide a minimum weight flange as well as the development of the leakage sensor.

b. Propellant and gas lines that are too small to utilize the dual seal design should be permanently joined where possible. When a mechanical joint is required, it could be wrapped with a chemically sensitive, color changing tape suited to the specific propellant. Each such joint must be inspected during ground operation. This approach provides inflight leakage data, but does not convey this information during flight. Rocketdyne is currently working on this technique.

c. Inflight monitoring of system and engine valve internal leakage could be accomplished by ultrasonic detection. It is not obvious that this technique can be developed to meet the probable leakage criteria. However, it is the only defined method that is potentially feasible and practical. Sensors implanted in the valve body near the internal seal offer promise for internal leakage detection providing that the requisite leakage signatures can be established.

d. The use of thrust chamber throat plugs in conjunction with a water displacement leak check has been used for some time to measure engine valve seal leakage. Since this is a very time-consuming operation, an effort should be made to circumvent this technique during the maintenance retest operation. If ultrasonic leak detection should not prove effective for direct detection, a thermal conductivity type leak detector could be considered for this application.

e. Regardless of the actual leakage detection methods used, an inflight hazardous gas monitoring system and an inert gas purging system will be required. The purge is required to expel possible hazardous gases from vehicle compartments. The hazardous gas monitoring system should be utilized to determine safe or unsafe conditions in the compartments. The data will also be used in evaluating ground maintenance requirements.

In general, leakage detection provisions must be incorporated into the design of the hardware. The dual seal concept and the implanting of sensors are examples of this requirement. The development of leak detection techniques should also be accomplished during the early stages of hardware development. Of particular benefit to the Space Shuttle program would be the development of leakage detection devices requiring no moving parts or complex mechanisms. It is therefore recommended that the polymeric gas detector, the UV spectrometer, the ultrasonic leak detector, color changing tapes and the thermal conductivity detector should be considered for further evaluation.

TABLE II-10

LEAK DETECTION CRITERIA BASELINE

LEAKAGE			MATCHING SENSITIVITY OF LEAK DETECTION METHODS
SOURCE/CONDITION	ROM (SCCS)*	REQUIREMENT	
Internal System Valves (Internal Leakage)	1×10^{-2}	Detect Quantify	Volumetric displacement, ultrasonics, thermal conductivity, radioactive tracer.
Engine Valve Seat (Seal Leakage)	1×10^{-2} to 1×10^{-3}	Detect Quantify	Volumetric displacement, thermal conductivity, radioactive tracer, halogen detector.
Engine Valves (External Leakage)	1×10^{-3}	Detect Quantify	Soap solution, chemical surface agents, thermal conductivity.
Check Valves (Reverse Flow Internal)	1×10^{-3}	Detect Quantify	Volumetric displacement, thermal conductivity.
Hazard Condition Monitoring (External Leakage)	$< 1 \times 10^{-3}$	Detect Locate	Hot filament, IR radiometer, catalytic reactor, UV spectrometer, thermal conductivity, gas analyzer, polymeric gas detector. Exposed sensor-Ion gas analyzer.
Mechanical Joints (External Leakage)	1×10^{-4} to 1×10^{-6}	Detect Locate Quantify	Thermal conductivity, halogen detector, mass spectrometer, chemical paper, radioactive tracer. Exposed sensor-Ion gas analyzer.
Mission Critical Joints (External Leakage)	1×10^{-7} to 1×10^{-9}	Detect Locate Quantify	Mass spectrometer, halogen detector, chemical paper, radioactive tracer.

*Rough order of magnitude in standard cubic centimeters per second.

TABLE II-11

LEAK DETECTION REFERENCES

CONTACTS:	SUBJECT
MARTIN MARIETTA CORPORATION	
Dr. James Bowman	Thermodynamic properties of H_2
Dr. J. A. Muscari	Leak detection methods
Mr. L. D. Williams	Leak detection methods
Mr. A. C. Anderson	Ultrasonic leak detection
Mr. N. H. Zaun	Spectroscopy
JET PROPULSION LABORATORY Mr. R. S. Weiner	Leakage Criteria
KENNEDY SPACE CENTER Mr. B. A. Tolson	Boeing H_2 Detection Study
LANGLEY RESEARCH CENTER Mr. V. L. Vaughn	Leak detection devices
DYNAMATEC CORPORATION Mr. J. C. Janus	Ultrasonic leak detection
ENVIRONMENT I INC. Mr. Robert Mack	Gas detection devices
GENERAL MONITORS, INC. Mr. K. Cobb	Hazardous gas detection
NORTH AMERICAN ROCKWELL CORPORATION Mr. Roy Dutton	Apollo leakage criteria

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Pearce, J. L. and Associates, Inc., Study to Develop Improved Methods to Detect Leakage in Fluid Systems. Phase I & II NAS10-7291 and NAS10-7510.

McDonnell Douglas Astronautics Co., Means of Leak Detection of Fluid Systems, Memo A 3-830-BH00-1107, 15 March 1971.

Air Force Rocket Propulsion Laboratory, Aerospace Fluid Component Designers' Handbook, February 1970.

Jet Propulsion Laboratory, Leakage Testing Handbook, NAS7-396, July 1969.

Aerojet Liquid Rocket Co., Investigation of TVC Lip-Seal Leak Check Test Methods, Internal Report, 24 October 1964.

McDonnell Douglas Astronautics Co., Study of Damage Control Systems for Space Station, NAS1-10184, October 1971.

Ohio University Research Institute, Leak Detection Technique Improvement Study for Space Vehicles, NAS8-11199, January 1967.

TABLE II-12
LEAKAGE EVALUATION RESULTS

COMPONENT/CONDITION	APPLICABLE TIME PHASE	RECOMMENDED LEAKAGE DETECTION METHOD
Hazardous Condition Monitoring (External Leakage)	Prelaunch Operations	A. Hot Filament Combustible Gas Detector B. Gas Analyzer C. Polymeric Gas Detector, UV Spectrometer
	Vehicle Ascent	A. Hot Filament Combustible Gas Detector B. Gas Analyzer C. Polymeric Gas Detector, UV Spectrometer
	Orbital Operation	A. Exposed Sensor-Ion Gas Analyzer B. Thermal Conductivity C. Polymeric Gas Detector, UV Spectrometer
	Reentry	A. Hot Filament Combustible Gas Detector B. Gas Analyzer C. Polymeric Gas Detector, UV Spectrometer
	Postflight Safing Operations	A. Hot Filament Combustible Gas Detector B. Gas Analyzer C. Polymeric Gas Detector, UV Spectrometer
Internal System Valves (Internal Leakage)	Maintenance/Retest	A. Volumetric Displacement *B. Pressure Decay C. Ultrasonics
	System Operation	A. — *B. System Pressures, System Dynamics C. Ultrasonics
	Quiescent Flight Periods	*A. System Pressures B. Ultrasonics C. Ultrasonics
Engine Propellant Valves (Internal Leakage)	Maintenance/Retest	A. Thermal Conductivity B. Volumetric Displacement C. Ultrasonics
	System Operation	A. N/A B. N/A C. N/A
	Quiescent Flight Period	A. — *B. Ultrasonics C. Ultrasonics
Mechanical Joints (≥2" dia) (External Leakage)	Maintenance/Retest	A. Thermal Conductivity B. Mass Spectrometer C. —
	System Operation	A. Thermal Conductivity *B. Capacitance Sensor C. —
	Quiescent Flight Period	A. Thermal Conductivity *B. Capacitance Sensor C.
Mechanical Joints (<2" dia) (External Leakage)	Maintenance/Retest	A. Mass Spectrometer B. Thermal Conductivity C. —
	System Operation	A. — B. Chemical Color Change Tape C. Polymeric Gas Detector
	Quiescent Flight Period	A. — B. Chemical Color Change Tape C. Polymeric Gas Detector

LEGEND: A. Prime leak detection candidate from available equipment or techniques that can be directly applied to the specified leakage condition.

B. The alternate choice to A. In all cases, the alternate is a secondary choice, but still has merit for the application considered.

C. Candidates which should be considered as a technical goal for the identified application. This type of candidate is not developed, but the capability has either been demonstrated or identified through available literature or contacts.

NOTE: When a candidate does not exist or an acceptable method is not known, a dash (—) has been substituted for the candidate choice.

*Does not fully meet the leakage detection requirements.

III TASK 2 - GUIDELINES DOCUMENT

III TASK 2 -
GUIDELINES DOCUMENT

A. OBJECTIVES

In our basic study, analytical techniques were developed and applied to define an approach for accomplishing the onboard checkout and monitoring functions of the Space Shuttle propulsion systems. Feasible approaches were defined for the functions of preflight checkout, performance monitoring, fault detection, fault isolation, emergency detection, postflight evaluation and maintenance retest.

With that background, the objective of Task 2 of the extension to Contract NAS8-25619 was stated in the Contract Scope of Work as follows:

> "Objective: To define and formulate a general specification for the incorporation of the onboard checkout and performance monitoring function into the design and development of the Space Shuttle Propulsion Systems. The specification shall define how to analyze the systems using a step-by-step approach. The specification shall be general in nature such that it can be imposed on the Space Shuttle contractors without regard to the detail designs of the vehicle hardware."

The ground rules that were established for the formulation of the document were:

- The document will be written as a General Specification in a specification format.

- Applicable documents will be included in the text of the specification where practical. Each document will be evaluated on an individual basis in this regard.

- Coordination with NASA will be accomplished for the topical outline, final outline, preliminary release, and final release of the general specification.

B. RESULTS

The original title selected for the document was "On-Board Checkout and Performance Monitoring Function for the Space Shuttle Propulsion Systems, General Specification for". The word "performance" was deleted from the title to preclude the sometimes narrow connotation that is associated with the phrase, "performance monitoring". The words

"propulsion systems" were deleted since the document was clearly applicable to other systems and equipment. As the writing of the document progressed, it became clear that it could be given even greater applicability and would get wider usage if the content were presented in a slightly less formal manner than is customary in a specification. For that reason, plus the fact that the document was to identify techniques and approaches rather than specify design, the words "general specification" were deleted in favor of "guidelines". Those changes and a little rearrangement led to the adoption of the title, "Guidelines for Incorporation of the Onboard Checkout and Monitoring Function on the Space Shuttle".

A general specification format has been followed in the document, however, the title of the customary "Quality Assurance" section has been deleted in favor of "Compliance Verification". This was done because the content of that section does not contain the traditional quality assurance material, but rather contains procedures and documentation requirements for the verification that the intent of the Requirements Section of the document have been complied with.

Since the applicable documents are small in number and are readily available from NASA and the Superintendent of Documents, U. S. Government Printing Office, Washington, D. C., 20402, they have been referenced rather than included in the body of the document.

For convenience, the guidelines document has been published under separate cover, as Volume II of this report.

IV TASK 3 - EXPENDABLE BOOSTER STUDY

A. OBJECTIVES AND RESULTS

The objective of Task 3 was to define and evaluate the propulsion checkout and monitoring requirements for the Titan III-L expendable booster. The task was conducted within the following guidelines:

1. Conclusions and recommendations were to be consistent with the objective of maximizing cost effectiveness in the Space Shuttle program.

2. Maximum use was to be made of the Space Shuttle Phase B Extension Study.

3. The study was restricted to the checkout and monitoring requirements of a selected baseline Titan III-L configuration.

4. The analyses were limited to typical system, subsystems, assemblies and components.

5. The task considered only the operational vehicle configuration.

6. Appropriate assumptions were made and documented where insufficient orbiter or booster data were not available.

The approach to Task 3 was based on the methodology developed during our basic study contract, and on the "Guidelines For Incorporation Of The Onboard Checkout and Monitoring Function On The Space Shuttle", Task 2 of the current study. The "Guideline" document was developed in parallel with this task, and the approaches and formats available during the development of each task element were incorporated.

The task was divided into four basic elements: Propulsion System Definition, Propulsion System Analyses, Checkout and Monitoring Requirements Analyses, and Checkout and Monitoring Requirements Implementation. This approach, which is in accord with the Task 2 methodology, has been utilized in the presentation of the material in the following sections.

The results and conclusions of the Titan III-L Checkout and Monitoring Requirements study are summarized as follows:

1. To maximize cost effectiveness, the checkout and monitoring techniques utilized on the operational Titan III's should be retained for the baseline Titan III-L configuration.

2. Certain control functions (flight control, booster staging, and portions of controls for emergency reaction) can best be accomplished from the orbiter.

3. The onboard checkout and monitoring functions identified in Task II are directly applicable to the expendable booster propulsion. Only the degree of usage and location of implementation are different.

4. The booster checkout and monitoring impact on the orbiter requirements is slight. A limited amount of data is transmitted to the orbiter for crew alert (caution and warning) and emergency action.

5. The impact on existing propulsion hardware design is minor. The direct impact is to the instrumentation, i.e., the number of required measurements, sensor redundancy, and sensor locations.

Two supporting research and technology items were identified, a solid rocket motor case burnthrough detector and a liquid rocket engine compartment fuel leakage or fire detector. Further evaluation of the best technical approach and of the current state-of-the-art are required before firm recommendations can be made on these items.

B. PROPULSION SYSTEM DEFINITION

This section defines the Titan III-L expendable booster propulsion systems that formed the basis for the study. First, a summary description of the baseline Space Shuttle vehicle configuration is presented on pages IV-3 and IV-4 . The booster main propulsion system (core) and the solid rocket motor system are then described. (For continuity with the subsequent analyses, the propulsion elements are defined in terms of systems, subsystems, assemblies and components, and a decimal nomenclature system is used.) Finally, the functional operations of the booster propulsions systems are discussed, starting on page IV-19.

1. Vehicle Configuration - The selected study configuration is the Martin Marietta Corporation Baseline Vehicle as of August 23, 1971, and is illustrated in Figure IV-1. The vehicle is

an expendable Titan III large diameter core booster (T III-L) and the Grumman H-33 drop tank orbiter. In this configuration, the orbiter is attached to the booster in a piggy-back fashion. The useful payload into a 100 nmi orbit is 45,000 lbs.

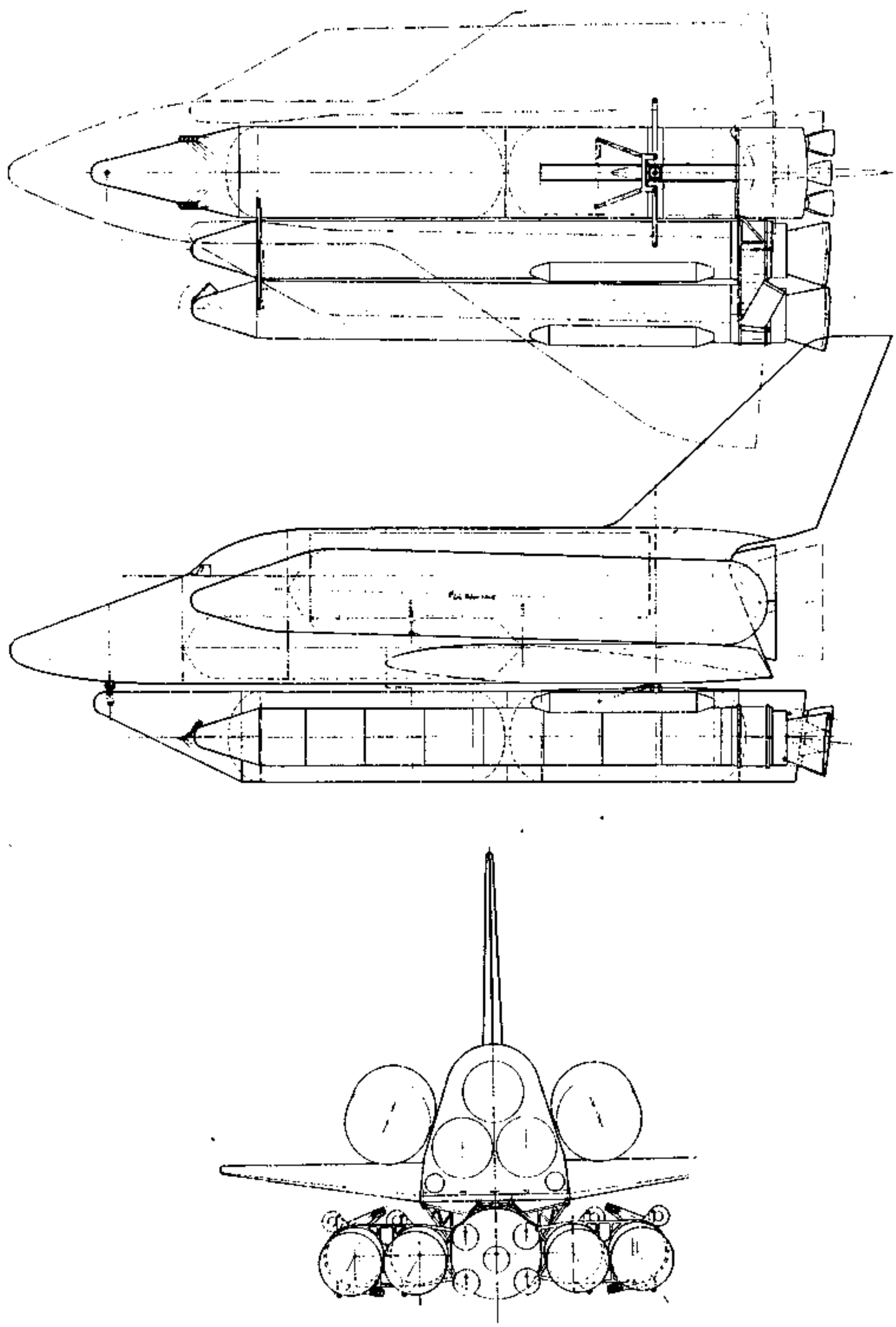


Figure IV-1 Vehicle Configuration

2. Booster Configuration - Figure IV-2 shows the Titan III-L booster, which is designated the 1207-4 spread configuration. The booster consists of a sixteen foot diameter liquid fuel core with five Aerojet Liquid Rocket Company LR-87 engines and four 120 inch United Technology Center UA 1207 solid rocket motors mounted in the yaw plane, two on either side of the core. The liquid-fuel core utilizes nitrogen tetroxide as the oxidizer and Aerozine-50, a solution of hydrazine and UDMH, as the fuel. The five core engines supply a total sealevel thrust of 1,133,370 pounds. The liquid rocket engines are precanted 9° in pitch to compensate for the Z-axis CG offset. The center engine is hinged in pitch to facilitate CG tracking, while the outboard engines have gimbal capabilities of $\pm$ 4.5 degrees in pitch and yaw. The four solid rocket motors develop a total of 5,570,400 pounds thrust. The solid rocket motor nozzles are precanted six degrees in a plane rotated 30 degrees out of pitch.

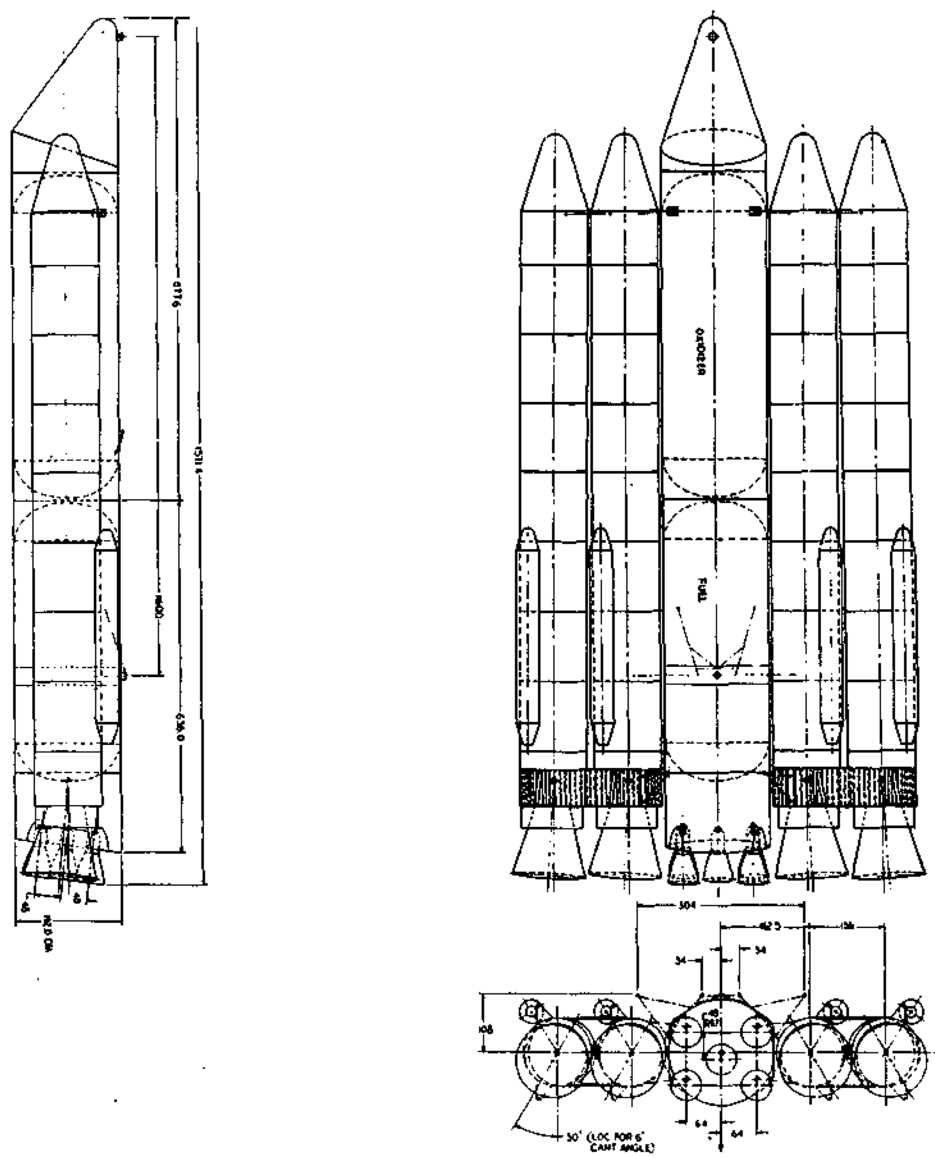


Figure IV-2 Booster Configuration

3. Booster Main Propulsion System (1.0) - The booster main propulsion system consists of the engines, propellant management subsystem, and pressurization subsystem on the liquid-fueled core of the Titan III-L booster. Figure IV-3 is a schematic representation of the system. Pertinent characteristics of the system are as follows:

Propellants	nitrogen tetroxide/50% hydrazine, 50% UDMH
No. of engines	five
Type	turbopump-fed
Thrust level	226,670 lb. each at sealevel
Ignition	hypergolic
Area ratio	12:1
Chamber pressure	800 psia
Thrust vector control	gimballed with hydraulic actuators
Oxidizer tank pressurization	autogenous (vaporized oxidizer)
Fuel tank pressurization	autogenous (cooled gas generator exhaust)
Usable oxidizer	800,600 lbs.
Usable fuel	419,000 lbs.
Burntime	276 seconds

The main propulsion system is further defined in the following pages.

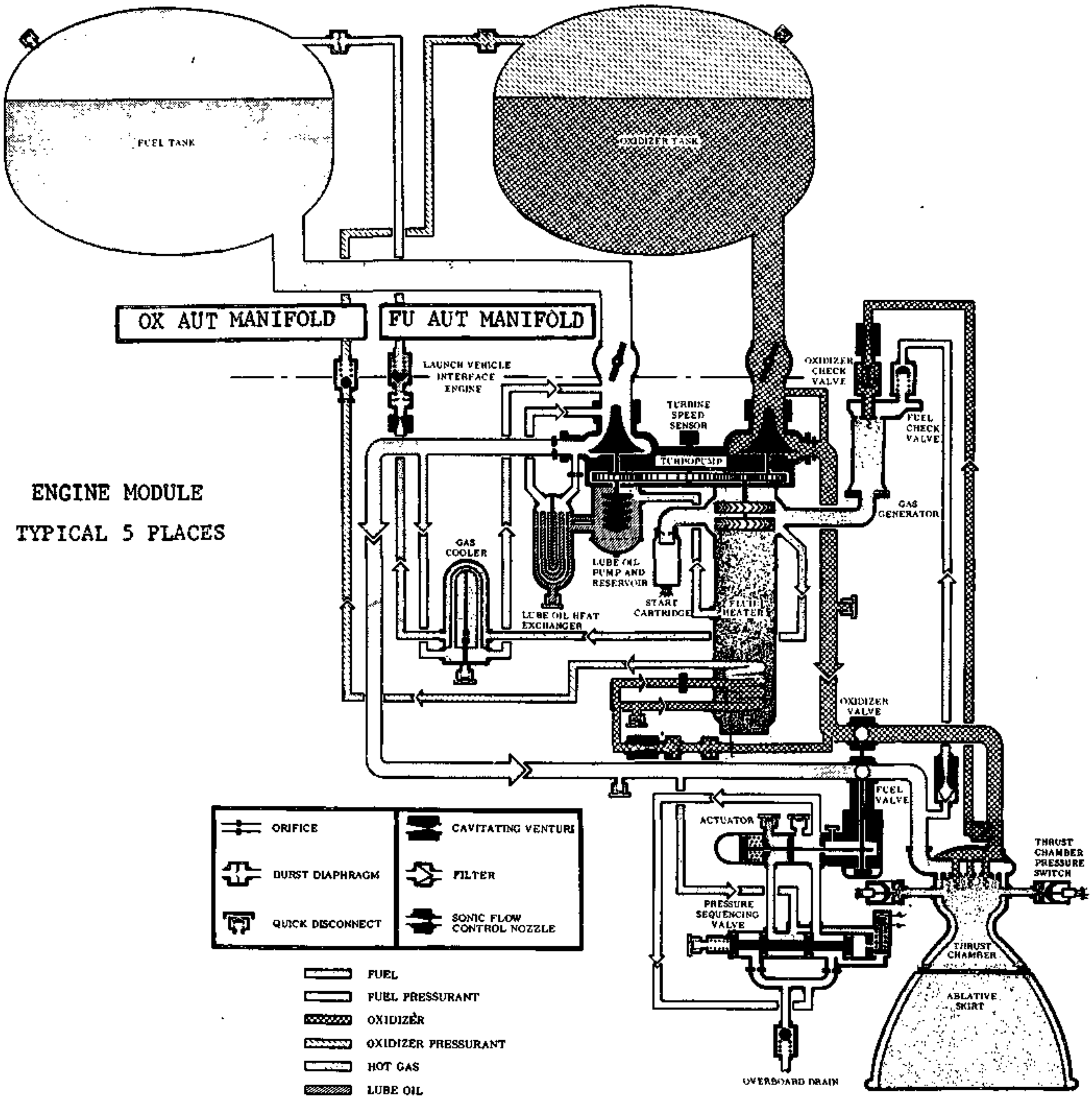


Figure IV-3 Main Propulsion System

a. Main Engine Subsystem (1.1) - The main engine subsystem is a storable liquid propellant, turbopump fed configuration that develops 226,670 lb sealevel and 257,750 lb altitude thrust. Five engines are used on the Titan III-L booster. The engine is shown schematically in Figure IV-3. The assemblies and components are identified in Table IV-1.

The suction lines duct the fuel (Aerozine-50) and oxidizer (nitrogen tetroxide) from the propellant feed lines to the pumps, which are driven by a turbine rated at 5000 h.p. The fluid pressure is increased through the pumps to force the propellants into the thrust chamber. Thrust chamber valves are utilized to control engine start and shutdown. The gas generator is operated by propellants from the pump discharge lines to drive the turbine. Combustion in the thrust chamber produces gas at a pressure of 800 psia and temperature of approximately 5000°F.

The engine requires no thrust or mixture ratio controls; it is pre-set to consume propellants at fixed rates. Balance orifices in the propellant discharge lines and cavitating venturis in the gas generator bootstrap lines determine the steady-state level. The propellant flow rate established by the discharge line orifices is a function of both upstream and downstream pressures. The cavitating venturis establish a flow rate that is sensitive only to upstream pressure, maintaining a constant flow rate over a wide range of downstream pressures. The control of propellant flow rate to the gas generator results in a stablized turbine speed. The propellants are hypergolic (they ignite on contact with one another), and, therefore, no ignition system is required to initiate combustion in the thrust chamber or gas generator.

Thrust vector control (pitch, yaw and roll) is achieved by pivoting the thrust chamber on gimbal bearing mounts. The gimbal action of the thrust chamber is provided by hydraulic actuators which operate in response to signals from the vehicle control system.

During booster operation, oxidizer and fuel tank pressurants are supplied by the engine subsystem. This part of the engine subsystem is discussed in the main pressurization subsystem section, page IV- 10.

TABLE IV-1
MAIN ENGINE SUBSYSTEM

- 1.1.1 Thrust Chamber Assembly
  - 1.1.1.1 Injector
  - 1.1.1.2 Combustion Chamber
  - 1.1.1.3 Ablative Skirt
  - 1.1.1.4 Thrust Chamber Pressure Switch (3)
- 1.1.2 Thrust Chamber Valve Assembly
  - 1.1.2.1 Oxidizer Thrust Chamber Valve
  - 1.1.2.2 Fuel Thrust Chamber Valve
  - 1.1.2.3 Pressure Sequencing Valve
  - 1.1.2.4 Overboard Drain Check Valve
- 1.1.3 Turbopump Assembly
  - 1.1.3.1 Oxidizer Pump
  - 1.1.3.2 Fuel Pump
  - 1.1.3.3 Gear Box
  - 1.1.3.4 Lube Oil Pump
  - 1.1.3.5 Lube Oil Heat Exchanger
  - 1.1.3.6 Turbine
  - 1.1.3.7 Exhaust Stack
- 1.1.4 Gas Generator Assembly
  - 1.1.4.1 Gas Generator
  - 1.1.4.2 Oxidizer Check Valve
  - 1.1.4.3 Fuel Check Valve
  - 1.1.4.4 Oxidizer Cavitating Venturi
  - 1.1.4.5 Fuel Cavitating Venturi
  - 1.1.4.6 Oxidizer Line Filter
  - 1.1.4.7 Fuel Line Filter
- 1.1.5 Engine Start Assembly
  - 1.1.5.1 Start Cartridge
  - 1.1.5.2 Initiator
- 1.1.6 Thrust Vector Control Assembly
  - 1.1.6.1 Gimbal Block
  - 1.1.6.2 Gimbal Actuator (2 on outboard engines, 1 on center engine)
  - 1.1.6.3 Hydraulic Pump
  - 1.1.6.4 Auxiliary Pump

b. <u>Main Propellant Management Subsystem (1.2)</u> - The main propellant management subsystem provides feed, distribution and storage of the propellants. The propellant tanks are mounted in tandem with the oxidizer tank in front. The fuel tank has 5 internal conduits to duct the oxidizer to the rocket engines. Both tanks have an access cover in the forward dome. The oxidizer tank is 16 feet in diameter and has a 9000 ft^3 volume. Five 7-inch diameter feedlines are used to deliver oxidizer to the engines. The fuel tank is also 16 feet in diameter with 7600 ft^3 volume. Five 6-inch diameter feedlines are used to deliver fuel to the engines. Appropriate tank baffles and contoured outlets are provided in each tank. Accumulators are used for POGO suppression.

TABLE IV-2
MAIN PROPELLANT MANAGEMENT SUBSYSTEM

1.2.1	Oxidizer Tank Assembly	
	1.2.1.1	Oxidizer Tank
	1.2.1.2	Oxidizer Pogo Suppressor (5)
	1.2.1.3	Oxidizer Prevalve (5)
1.2.2	Fuel Tank Assembly	
	1.2.2.1	Fuel Tank
	1.2.2.2	Fuel Pogo Suppressor (5)
	1.2.2.3	Fuel Prevalve (5)

c. Main Pressurization Subsystem (1.3) - The main pressurization subsystem provides the necessary pump suction pressure for proper pump operation. The propellant tanks are pre-pressurized with nitrogen prior to engine start through the two-inch tank vents. During engine operation, pressurizing gas is supplied to the tanks by the engine autogenous (self-generating) system at a controlled rate to make up for the removal of propellant from the tanks. The fuel tank is pressurized by diverting a portion of the engine gas generator output from the turbine inlet manifold to the tank. This gas is cooled by passing it through a fuel-coupled heat exchanger. Fuel flows from the pump discharge line through the gas cooler back to the suction side of the pump. A sonic nozzle in the autogenous gas line maintains a flow rate that is insensitive to tank pressure, and a 300 psid burst diaphragm prevents gas flow to the tank until gas generator operation begins.

The oxidizer tank is pressurized by heating oxidizer to the gaseous state and ducting it to the tank. Oxidizer is piped from pump discharge to a heat exchanger (fluid heater) located in the turbine exhaust stack. A cavitating venturi, located at the fluid-heater inlet, maintains a constant flow rate insensitive to downstream pressure, and a burst diaphragm prevents oxidizer autogenous flow until the discharge pressure reaches approximately 300 psia. A loop in the inlet line traps air to ensure pneumatic operation of the burst diaphragm. A back pressure orifice provides sufficient residence time of the oxidizer in the fluid heater to achieve the proper gas temperature of the pressurant.

The Main Pressurization Subsystem is shown schematically in Figure IV-4, and the assembly and component identification is presented in Table IV-3.

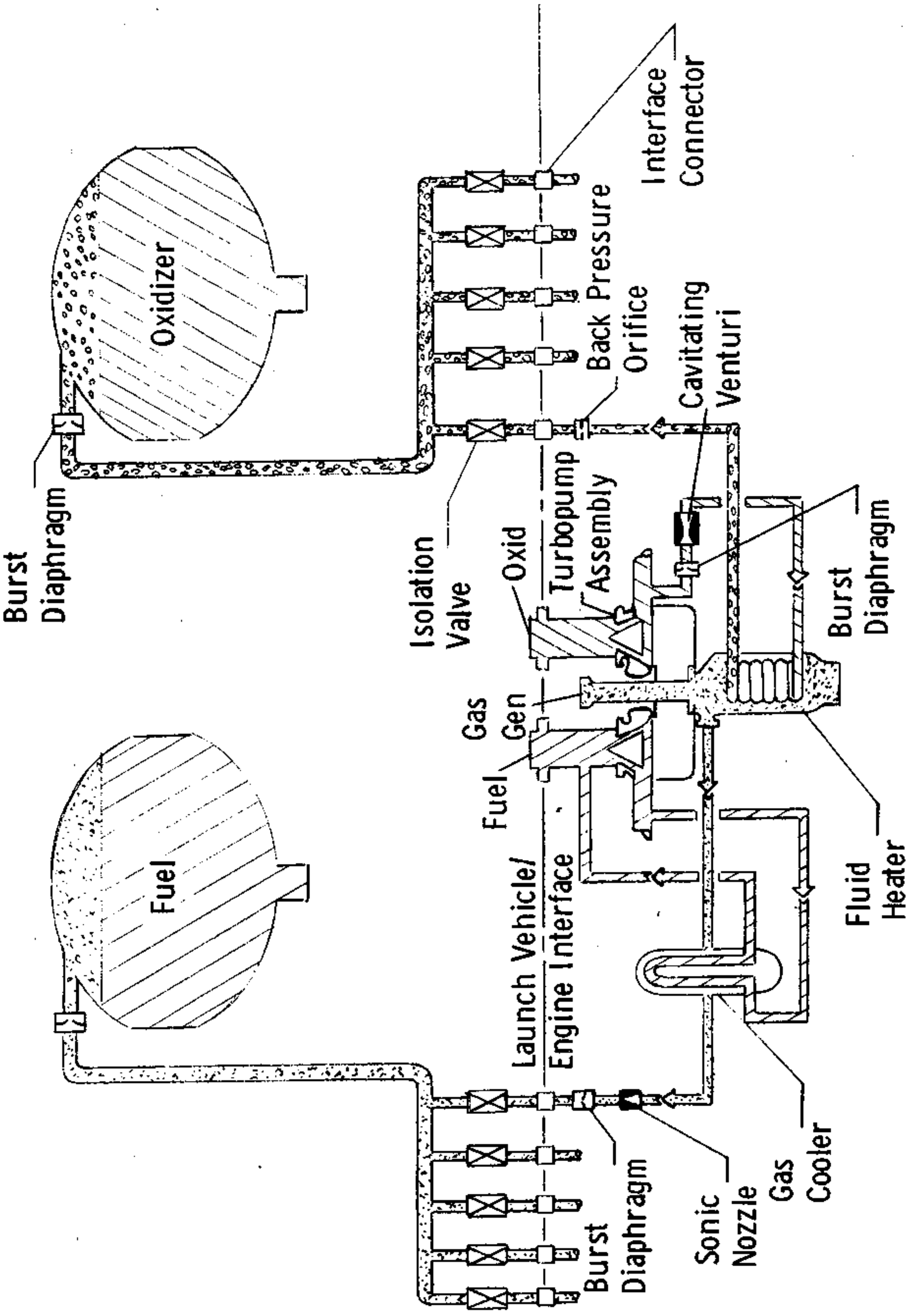


Figure IV-4 Main Pressurization Subsystem

TABLE IV-3
MAIN PRESSURIZATION SUBSYSTEM

1.3.1 Oxidizer Pressurization Assembly

- 1.3.1.1 Fluid Heater
- 1.3.1.2 Cavitating Venturi
- 1.3.1.3 Back Pressure Orifice
- 1.3.1.4 Burst Diaphragm (autogenous)
- 1.3.1.5 Check Valve
- 1.3.1.6 Oxidizer Pressurization and Vent Valve
- 1.3.1.7 Oxidizer Tank Diaphragm

1.3.2 Fuel Pressurization Assembly

- 1.3.2.1 Gas Cooler
- 1.3.2.2 Sonic Nozzle
- 1.3.2.3 Burst Diaphragm (autogenous)
- 1.3.2.4 Check Valve
- 1.3.2.5 Fuel Pressurization and Vent Valve
- 1.3.2.6 Fuel Tank Diaphragm

4. Solid Rocket Motor System (2.0) - The solid rocket motor system consists of four solid rocket motor (SRM) and thrust vector control (TVC) subsystems. Each SRM consists of a forward closure, an aft closure, and identical, interchangeable segments. Other components include a 6° canted nozzle, and fore and aft solid propellant staging rockets. The TVC injectant, nitrogen tetroxide, is carried in a tank mounted on the side of the SRM and is pressure fed into the nozzle exit section by nitrogen gas. Figure IV-5 illustrates components and assemblies of the solid rocket motor system.

   a. Rocket Motor Subsystem (2.1) - The selected motor is composed of seven segments and is designated as Model 1207. Thrust termination capability, developed on the original T-IIIC SRM, is provided for Model 1207 through the use of ports, located at the forward end of the motor, which may be ejected on command to permit thrust termination in case mission abort becomes necessary.

   The motor case (segments and closures) is constructed of D6aC steel, heat-treated to an ultimate strength of 195,000 psi. Each joint is a pin and clevis type held together by 240 cylindrical pins. During assembly of the motor, the pins are inserted by hand and held in place by a retaining strap. A gas pressure seal between segments (and closures) is provided by an O-ring.

   Each segment contains approximately 72,400 lb. of polybutadiene acrylic acid acrylonitrile (PBAN) composite propellant which uses powdered aluminum fuel and ammonium perchlorate oxidizer. The plastic matrix, PBAN, also serves as a fuel. The case-bonded propellant grain has a circular port which tapers 10 inches throughout the 10-foot length of the segment. The forward end has the smaller port. The purpose of this taper is to provide the 10-second controlled tail-off at the end of web-action time. The forward end of each segment is inhibited from burning by a rubber restrictor bonded to the propellant surface. Silica-filled, butadiene acrylonitrile rubber insulation protects the motor case from combustion gas during motor operation. The insulation is thickest in the segment joint areas where there is no unburned propellant to protect the case walls.

   The closures contain the same type of propellant as the segments, and the forward closure has mounting provisions for the igniter. Instead of the cylindrical grain shape of the segment, the forward closure has an 8-point star internal burning grain configuration. The forward closure is 135 inches long and contains 61,000 lb. of propellant.

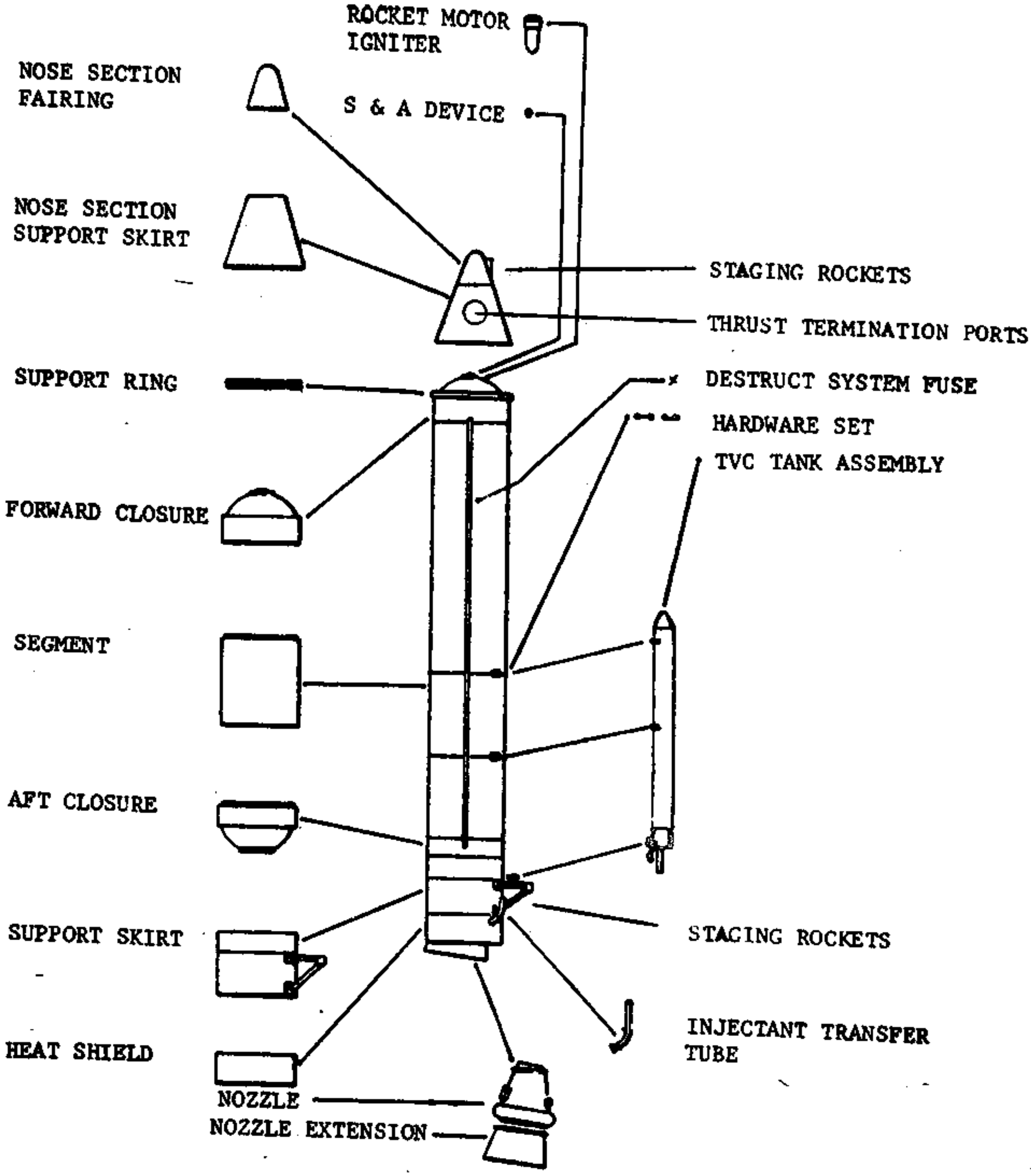


Figure IV-5 Solid Rocket Motor System

The aft closure contains approximately 20,300 lb. of propellant in a straight cylindrical bore configuration and projects 64 inches from the segment joint to a 57-inch diameter boss for nozzle attachment.

The propellant burns along the entire central port of the SRM and also on the aft end of each segment between segments. The closures also burn on their ends. Three inches of clearance are left between the grains of adjacent segments to permit this burning. The igniter burns for approximately 1 second to fill the grain bore with burning gases to ignite the motor. The SRM has a regressive thrust-time curve produced in part by the star configuration of the propellant grain in the forward closure of the motor. During the early phases of burning, this portion contributes much of the gas flow necessary to produce the high initial peak in the thrust-time curve.

The SRM nozzle consists of a throat section and a two-piece exit cone assembly. High-density graphite rings backed by a steel support shell and silica insulation are bonded in a steel housing to make up the nozzle throat section. The nozzle middle section consists of graphite and silica phenolic liners bonded to a steel outer shell. This section contains the thrust vector control injection ports. The exit section is an extension of the silica phenolic liner of the middle section except that its structural shell is aluminum honeycomb sanwiched between steel for lighter weight. The three sections are bolted together, forming an assembly approximately 14.5 feet long. Nozzle expansion ratio is 9.18:1, and the half-angle is 17°. The SRM assemblies and components are listed below. Figure IV-6 depicts the subsystem.

TABLE IV-4
ROCKET MOTOR SUBSYSTEM

2.1.1	SRM Assembly	
	2.1.1.1	Forward Closure
	2.1.1.2	Segment (7)
	2.1.1.3	Aft Closure
	2.1.1.4	Nozzle
	2.1.1.5	Rocket Motor Igniter
	2.1.1.6	Thrust Terminating Device
	2.1.1.7	Destruct Device
2.1.2	Staging Rocket Assembly	
	2.1.2.1	Staging Rocket Motor (18)
	2.1.2.2	Staging Rocket Motor Housing

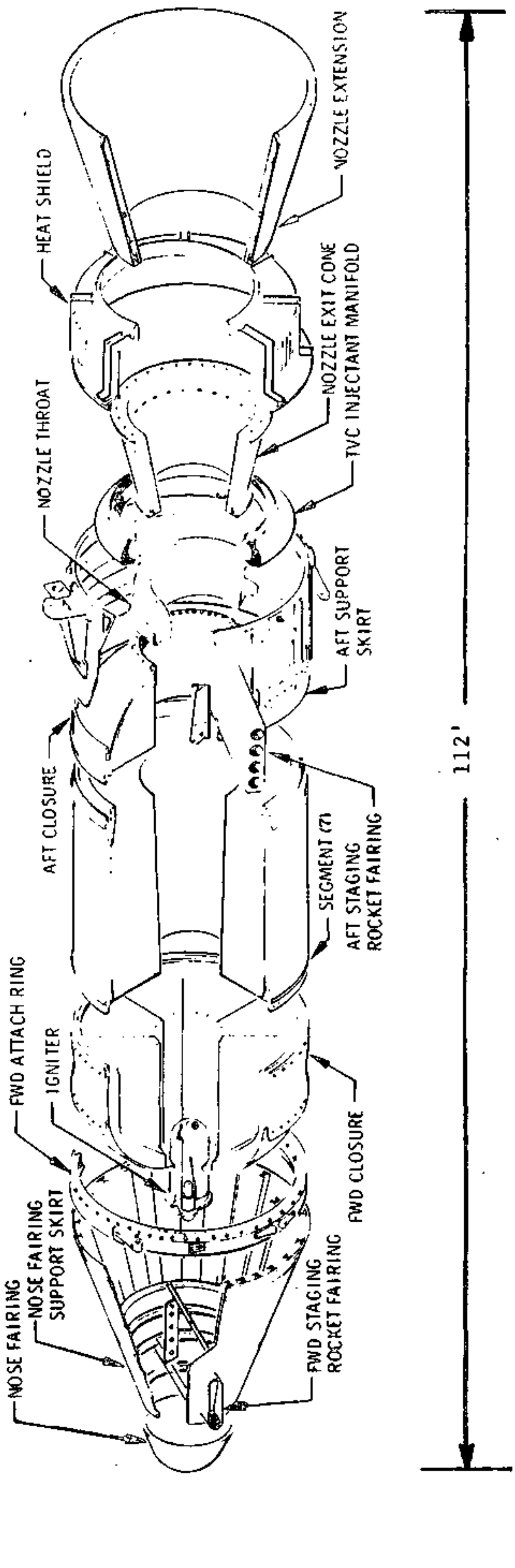


Figure IV-6 Rocket Motor Subsystem

b. Thrust Vector Control Subsystem (2.2) - Thrust vector control for the UA 1207 SRM is achieved through fluid injection in the nozzle exit cone. The 3.5 ft. diameter injectant tank, which is mounted on the side of the SRM, contains 12,000 lb. of nitrogen tetroxide and is charged with nitrogen gas to a pressure of 1,100 psi. The ullage pressure decays to about 500 psi during the flight duty cycle.

Nitrogen tetroxide passes through an injectant transfer tube and a toroidal injectant manifold to the 24 electromechanical injectant valves. These valves are actuated electrically in response to commands from the core flight control system. The TVC subsystem can provide the required side forces with only five of the six valves operable in any quadrant. The subsystem provides a jet deflection of 0 to 4 degrees. The subsystem is designed to dump excess nitrogen tetroxide during booster operation to minimize burnout weight, and is capable of a slew rate of 10 degrees per second at all times during SRM operation.

During countdown, pressurized fluid fills the entire TVC system through the injectant valves to aluminim cylinders and caps (pyroseals) that protrude into the nozzle exit cone. The pyroseals and caps extend a short distance into the nozzle and are burned off at ignition by the exhaust flame, thus activating the TVC system. The subsystem configuration is shown in Figure IV-7. The TVC assemblies and components are as follows:

TABLE IV-5
THRUST VECTOR CONTROL SUBSYSTEM

2.2.1	TVC Tank Assembly	
	2.2.1.1	Injectant Tank
	2.2.1.2	Nitrogen Pressurization Valve
	2.2.1.3	Injectant Fill and Drain Valve
	2.2.1.4	Injectant Transfer Tube
2.2.2	Injectant Valve Housing Assembly	
	2.2.2.1	Injectant Valve (24)
	2.2.2.2	Pyroseal (24)

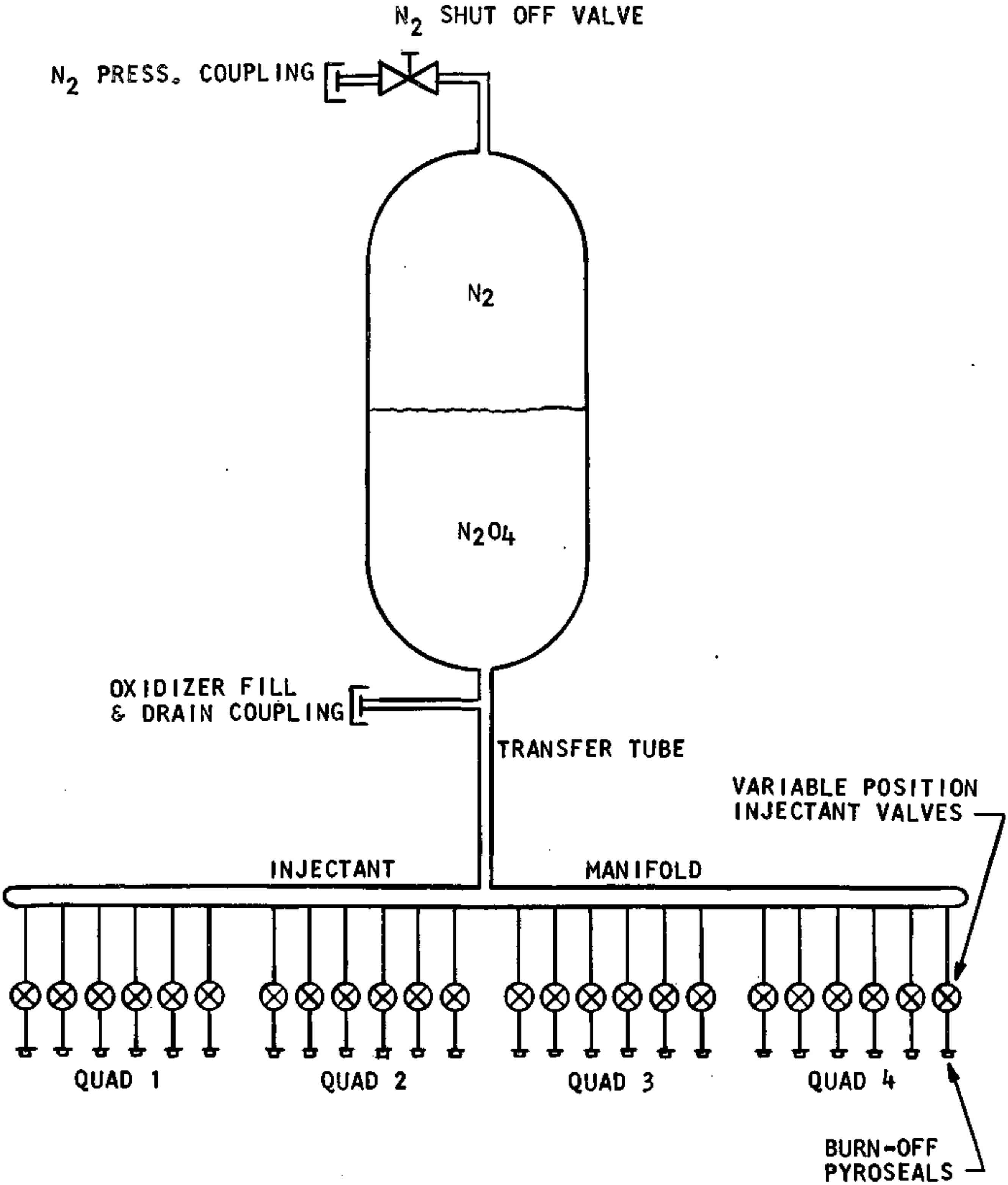


Figure IV-7 Thrust Vector Control Subsystem

5. Operations - Figure IV-8 depicts the operational timeline for the Titan III-L and the drop tank orbiter. The booster vehicle checkout and assembly is similar to the operation presently performed on the Titan III-B vehicle. A checkout matrix is shown in Table IV-6 and the vehicle assembly flow is shown in Figure IV-9. The main propulsion tanks are loaded with 800,600 lb. oxidizer and 419,000 lb. fuel. The TVC tank is loaded with 12,000 lb. of nitrogen tetroxide. At T-60 hours, the main propellant management subsystem is pre-pressurized to 28 psia and the TVC subsystem is pressurized to a flight pressure of 1,100 psia. At T-2 hours, after arming and ready status verification, the final countdown is initiated. At T-35 seconds, an automatic sequence signal is supplied to the launch vehicle. This signal accomplishes the following: (1) the prevalves are opened permitting fuel and oxidizer to fill the engine, and (2) the engine starting electrical circuits are readied for receipt of the firing signal.

Opening the prevalves places the engine in the fill and bleed condition. Both fuel and oxidizer fill the engine above the thrust chamber valves due to the static pressure of the propellants in the tanks above the engine. Air entrapped in the oxidizer lines travels through 3/8 inch flex lines, connected on each subassembly from the discharge line near the pump outlet flange (high point) to the suction line, up into the oxidizer tank. Air removal from the fuel lines is accomplished as fuel hydraulic pressure actuates the thrust chamber valves at engine start.

The fuel-operated valve actuation system consists of a rod and piston mechanically linked to the thrust chamber valves (TCVs), held closed by springs and opened by fuel pressure. A pressure sequencing valve (PSV), also held closed by a spring and opened by fuel pressure, acts as a pilot valve to the TCV actuator. Fuel bleed is accomplished by allowing fuel to flow through a 1/2 inch flex line from the high point on the discharge line at the TCV discharge line connecting flanges to the PSV inlet port. While in the bleed position, the PSV diverts the fuel into and through the closing side of the TCV actuator, through a 1/4 inch stainless steel vent line to an overboard manifold mounted on the PSV, and out an overboard drain line through a check valve which serves only to protect the PSV from contamination. A bleed orifice, located in the drain line and PSV manifold connection, controls the bleed rate to approximately 1200 cc per minute per engine. As long as the engine remains in the fill and bleed condition, fuel is bled overboard in the manner and at the rate described.

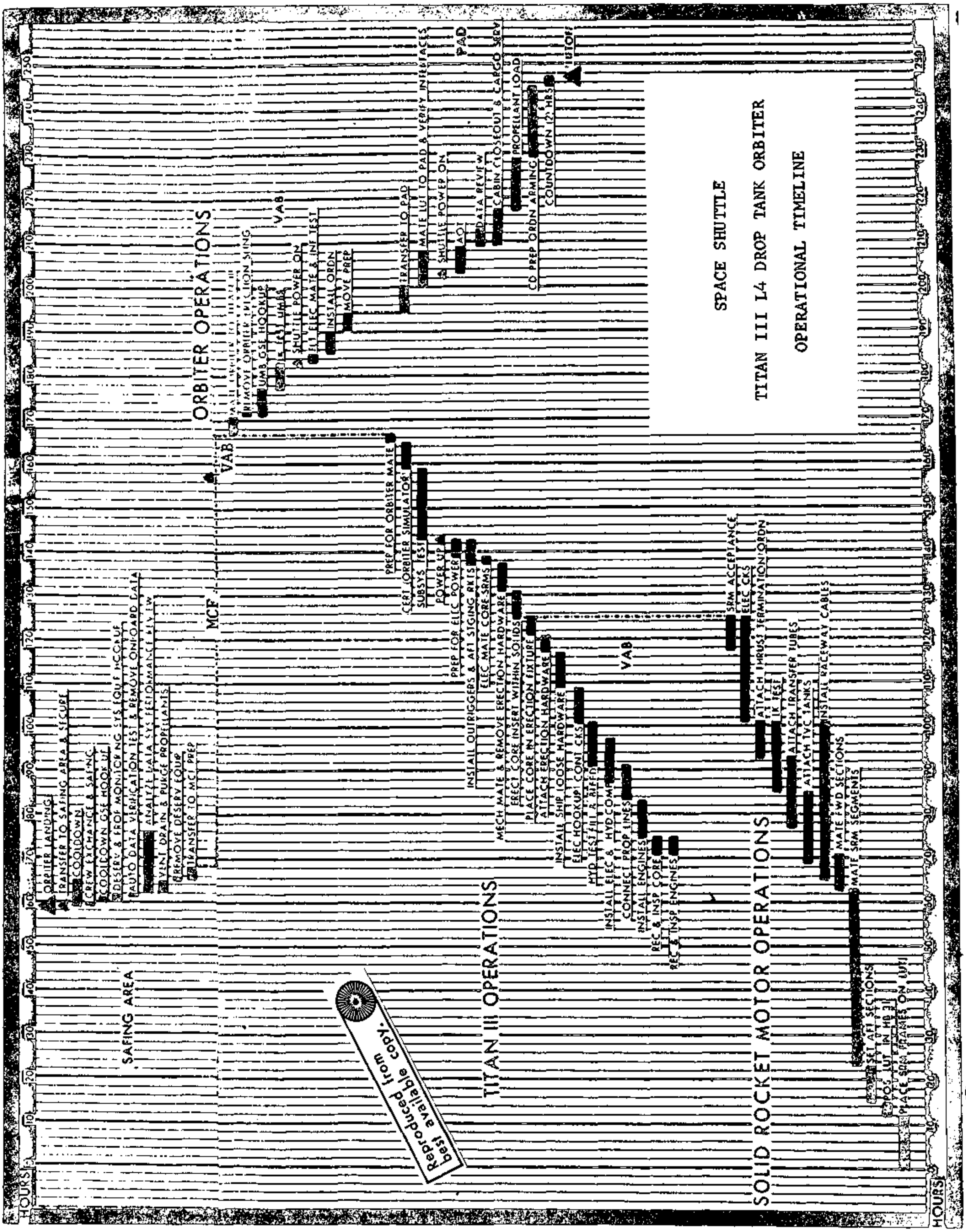


Figure IV-8 Operational Timeline

TABLE IV-6

BOOSTER PROPULSION CHECKOUT

Test Nomenclature	Supplier Facility	Prep Area	Launch Pad
1. Booster Main Propulsion System			
Instrumentation Check	X	X	X
Electrical Harness Check	X	X	
TCPS Check	X	X	
TPA Torque Check	X	X	
Thrust Chamber Valve Functional	X	X	
PSVOR Position Verification	X	X	
Torque Verification Test	X	X	
Engine Leak Checks	X	X	
Ordnance Test	X	X	
Propellant Tank Hydrostat	X		
Propellant Tank Leak Check	X	X	X
Prevalve Switch Test	X		
Propellant Tank Calibration	X		
Prevalve Boroscope Inspection	X	X	
Accumulator Compliance Test	X		
2. Solid Rocket Motor System			
Instrumentation Check	X	X	
Alignment Checks	X	X	
TVC Subsystem Leak Check	X	X	
Injectant Valve Functional	X	X	
Ordnance Test	X	X	
Composite System Test	X	X	
SRM Leak Check		X	
3. Combined Propulsion System			
Flight Instrumentation Check		X	X
Ordnance Check		X	X
Combined System Leak Check		X	
Combined System Test		X	X

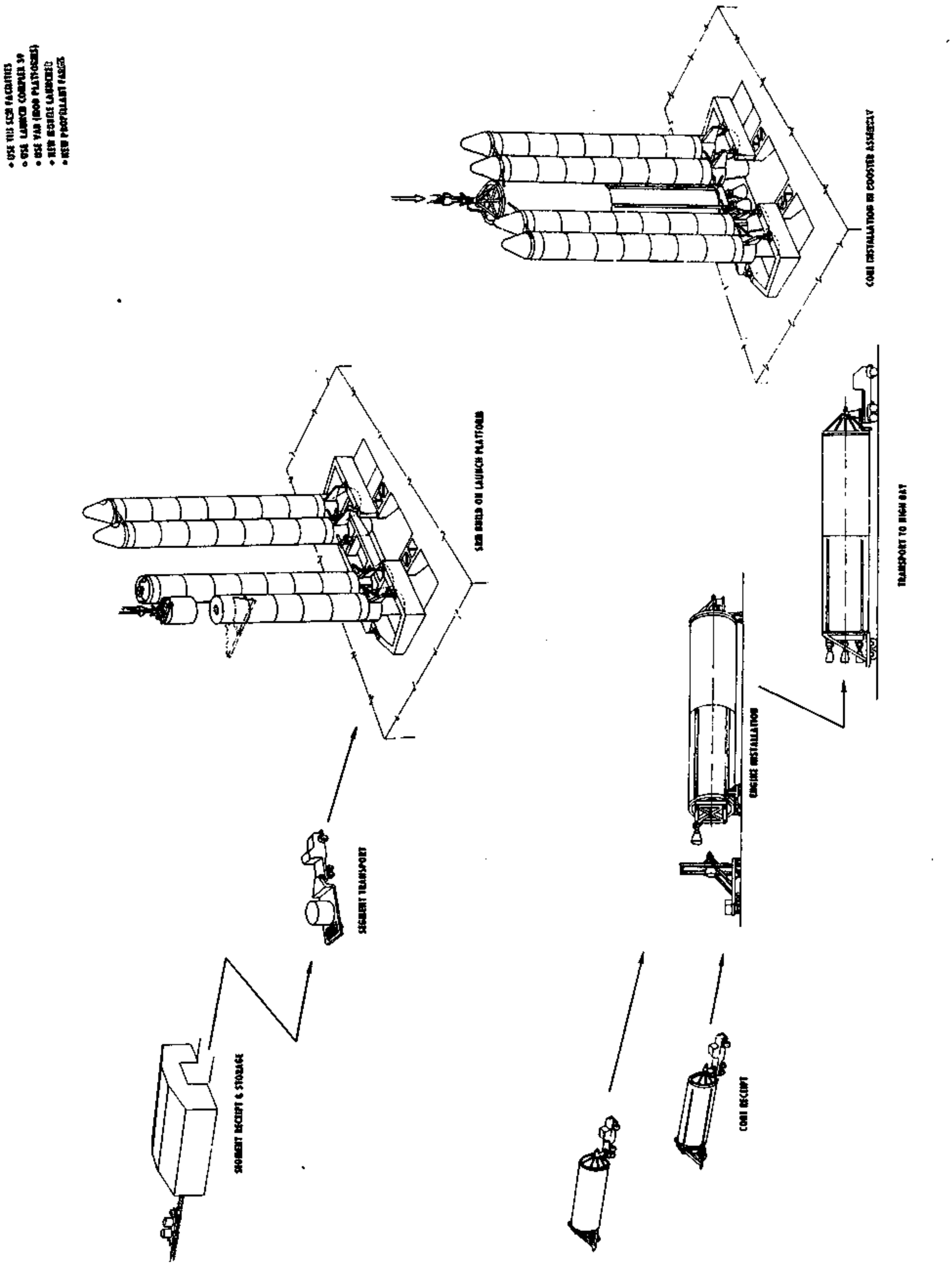


Figure IV-9 Vehicle Assembly Flow

After completing the bleed cycle, the start signal applies 28 vdc to the initiator charges of a solid propellant start cartridge mounted on the turbine inlet manifold of each engine. Two diametrically opposite engines are started sequentially at 0.1 second intervals. The center engine is started at 0.1 second after start command to the last pair. The start cartridge solid propellant ignites and supplies gas to the turbines causing them to accelerate. The turbine shaft of each engine is connected through a gear train to the fuel and oxidizer pump so that pump operation also begins. Since the thrust chamber valves are closed, no propellant flows, and pump acceleration produces only an increasing pressure in the discharge lines and valve actuation system.

When fuel discharge pressure reaches approximately 300 psia, the pressure on the opening end of the PSV spool produces a force which exceeds the spring force on the PSV spool closing end causing the spool to shuttle from the bleed position to the operation position. This occurs approximately 0.25 seconds from start signal. Relocation of the PSV spool to the operation position halts flow of fuel to the closing side of the TCV actuator, terminating fuel bleed, and allows fuel to enter the actuator opening side. The fuel pressure, increasing beyond the 300 psia PSV actuation pressure, is immediately sufficient to operate the TCV actuator, initiating opening of the thrust chamber fuel valve. The fuel and oxidizer valves are mechanically linked so that both move simultaneously. The rate of motion of the TCV actuator piston is controlled by an orifice located in the overboard drain manifold to PSV housing connection nearest the PSV opening end. This opening orifice controls the rate at which bleed fuel can be expelled from the TCV actuator. Valve opening begins approximately 0.3 second after start cartridge initiation. At approximately 1.1 seconds after start signal the valves are fully opened.

Propellants begin to flow to the thrust chamber at the time the valves begin to open. Oxidizer flows directly into the injector dome, filling the oxidizer injector manifold, through orifices into the combustion chamber. Fuel is used to regeneratively cool the combustion chamber and must fill a toroidal manifold and flow through stainless steel cooling tubes that make up the chamber walls before reaching the injector fuel orifices and entering the combustion zone. The larger volume the fuel must fill before reaching the injector orifices results in an oxidizer lead into the combustion zone. This slight oxidizer

lead (0.25 to 0.30 sec) provides a characteristically smooth start. Initial pressurization of the chamber during oxidizer lead ejects the skirt exit closures.

Opening the thrust chamber valves causes a momentary decrease in discharge pressures as the volume between the TCVs and injector orifices is being filled with propellant. At approximately 0.8 second after start signal this filling is completed and hypergolic ignition of the fuel and oxidizer takes place in the combustion chamber. At this point in time pressure in the discharge lines again increases, and fuel and oxidizer are forced through flex lines attached from the main propellant discharge lines, downstream of the TCVs, to the gas generator. Check valves located at the gas generator end of these bootstrap lines prevent the flow of start cartridge gas into the discharge lines. The gas generator ignites at approximately 0.9 second after start signal and supplies gas to drive the turbines. At approximately 1.1 seconds after start signal, the start cartridge burns out, and the engine has reached its operating level, i.e., it has "bootstrapped".

The outputs of the three thrust chamber pressure switches (TCPSs) of each liquid engine are majority voted to provide an engine operation status signal. The TCPSs are calibrated to switch when engine chamber pressure has attained 77% of nominal operating pressure. The receipt of nominal chamber pressure indications from all five liquid engines is a condition to initiate the solid rocket motor ignition sequence. The SRM ignition command fires redundant squibs located in the Safe and Arm device of each SRM. A pyro train provides the required energy to ignite the main propellant grain, providing a nominal thrust build up in 250 msec. The grain is designed to produce an initial thrust of 1.5 million pounds (which regresses to approximately 1.1 million pounds in 115 seconds, followed by a 15 second tail-off.)

The nozzle operating temperature is more than sufficient to burn off the aluminum "pyro-seal" closures at the outlet of each TVC injectant valve, thereby activating the thrust vector control subsystem. Side forces of up to 100,000 pounds are provided to each SRM by the thrust vector control subsystem on command signals from the core. Pressurized N_2O_4 is injected into the nozzle exit cone through six electromechanical valves in each quadrant which deflect the exhaust gases through the formation of an oblique shock wave in the SRM nozzle.

Shortly after end of web action time, sensed deceleration initiates the SRM staging timer. At a preprogrammed time interval, the forward attach explosive nuts, the aft explosive attach bolts, and the staging rockets are activated. The SRMs are staged simultaneously in pairs such that the resultant direction is down and away from the core vehicle.

At T+130 seconds a core-orbiter overlap burn is initiated by starting two of the three orbiter main engines. Depletion of either or both of the core propellants commands LRE shutdown and orbiter staging at approximately T+277 seconds.

Exhaustion of either or both core propellants is determined by the liquid engine TCPs which are monitored during an appropriate time interval (TCPS enable). The core engine shutdown sequence is initiated when the first of the five TCPS signals indicates that engine chamber pressure has decreased to 77% or less of nominal operating pressure. As in the engine start sequence, each of the five TCPS signals is a resultant of the indications of three majority voted pressure switches.

At T+282 seconds, the third orbiter main engine is started. The orbiter attains orbit inject velocity at approximately T+444 seconds. The sequence of major events from propellant loading to orbit inject is presented in Table IV-7. The ascent trajectory profile is shown in Figure IV-10.

TABLE IV-7

SEQUENCE OF EVENTS

TIME	EVENT
T-72 Hrs	Complete propellant loading.
T-60 Hrs	Pressurize tankage.
T-12 Hrs	Ordnance arming.
T-8 Hrs	Final pressurization adjust.
T-195 Min	Start final countdown
T-35 Sec	Automatic sequence start. (terminal count)
T-34 Sec	Initiate prevalve opening.
T-3.0 Sec	Start LRE ignition sequence.
T-0.25 Sec	Initiate SRM ignition.
T-0	Liftoff.
T+5.6 Sec	Vehicle clear of pad structure.
T+126 Sec	SRM burnout and staging.
T+130 Sec	Start two orbiter main engines.
T+277 Sec	Booster LRE shutdown and booster staging.
T+282 Sec	Start Orbiter No. 3 engine.
T+444 Sec	Orbit inject.

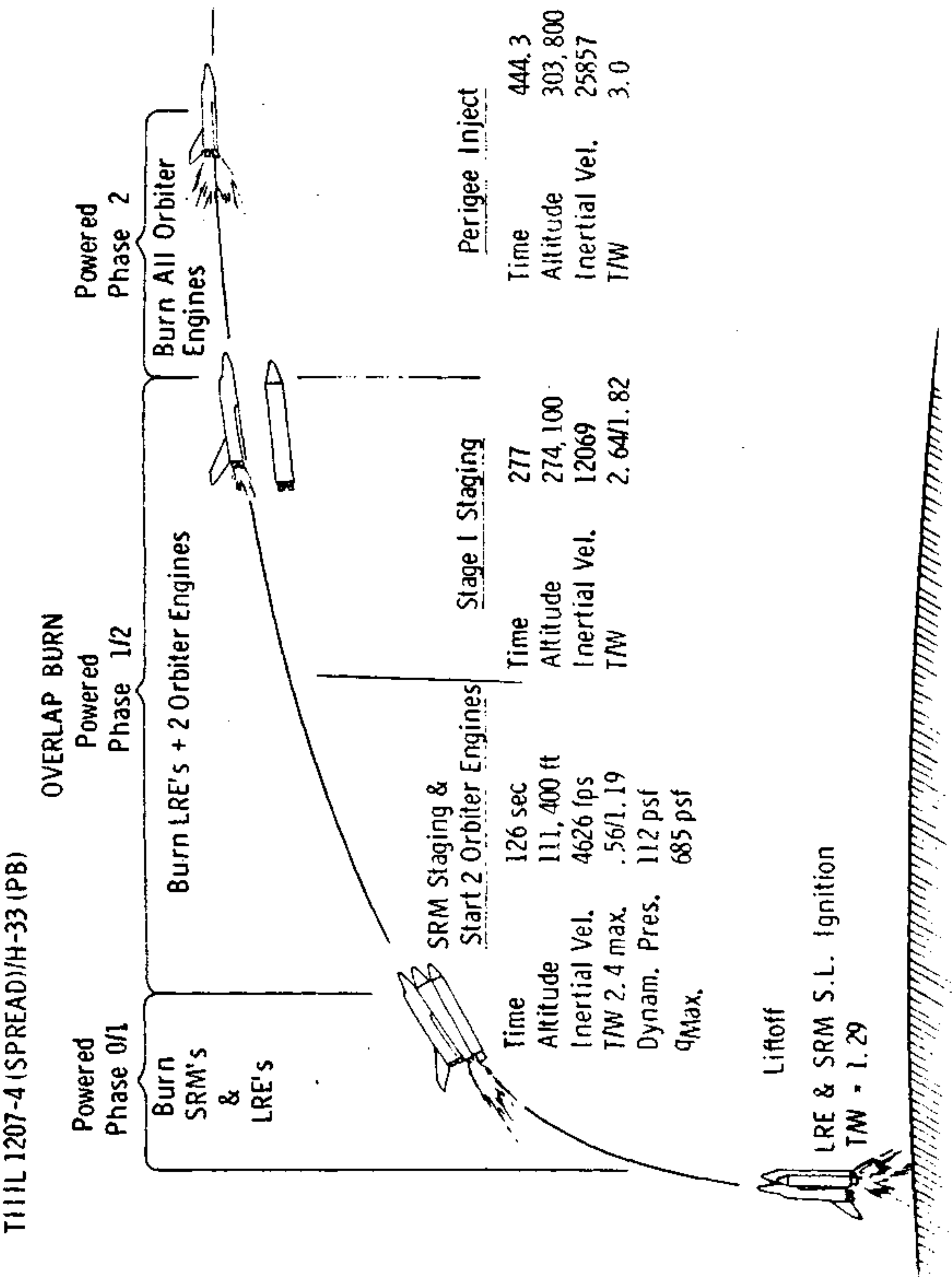


Figure IV-10 Ascent Trajectory Sequencing Profile

6. Propulsion-related Avionics - The avionics related to the propulsion systems consists of portions of the vehicle electrical and electronic subsystems. An overall view of the interconnections between propulsion and avionics is shown in Figure IV-11. These interconnections are discussed in the following paragraphs.

   a. Digital Interface Units (DIUs) - The DIUs are extensions of the orbiter Central Computer Complex (CCC), and provide the interfaces between the booster avionic subsystems and the CCC via the digital data bus. Propulsion related signals which interface with the DIUs, depicted in Figure IV-12, are as follows:

      1) Steering and Dump signals are provided from the DIU to the booster thrust vector control (TVC) driver electronics. The steering signals are issued in accordance with the desired flight attitude which is controlled by the orbiter. The dump signals are issued from the orbiter in accordance with a nominal preprogrammed dump schedule. The TVC electronics interprets the steering and dump signals and provides the outputs required to drive the actuator valve coils in the liquid rocket engines and the injectant valve positioning electronics in the solid rocket motor TVC subsystem. The TVC driver electronics integrates the outputs to the SRM valves to determine the total injectant fluid usage versus flight time and adjusts the SRM steering-dump commands to correspond with the preprogrammed dump schedule.

      2) Rate and lateral acceleration inputs from booster flight control components are sent to the orbiter via the DIU. The booster rates and accelerations are used in conjunction with orbiter attitudes and rates to determine the steering required.

      3) Malfunction detection logic outputs from the booster are provided to the orbiter CCC via the DIU. The malfunction detection logic outputs provide the status of booster parameters which can indicate that a hazardous condition exists on the booster.

      4) Discrete sequencing and driving circuit inputs from the DIU and malfunction detection logic are provided to control LRE and SRM functions such as start and shutdown commands.

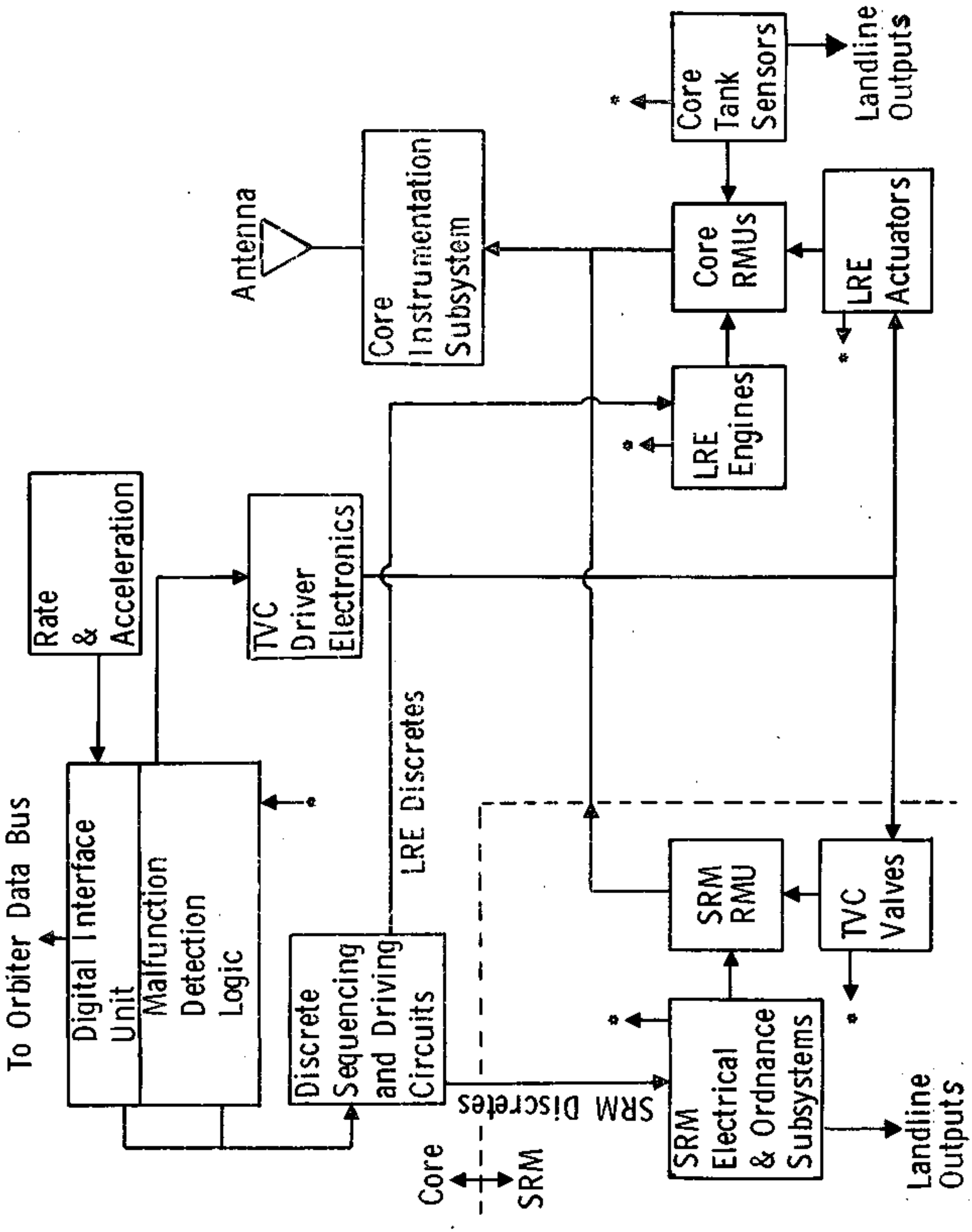


Figure IV-11 Titan III L Propulsion Related Avionics

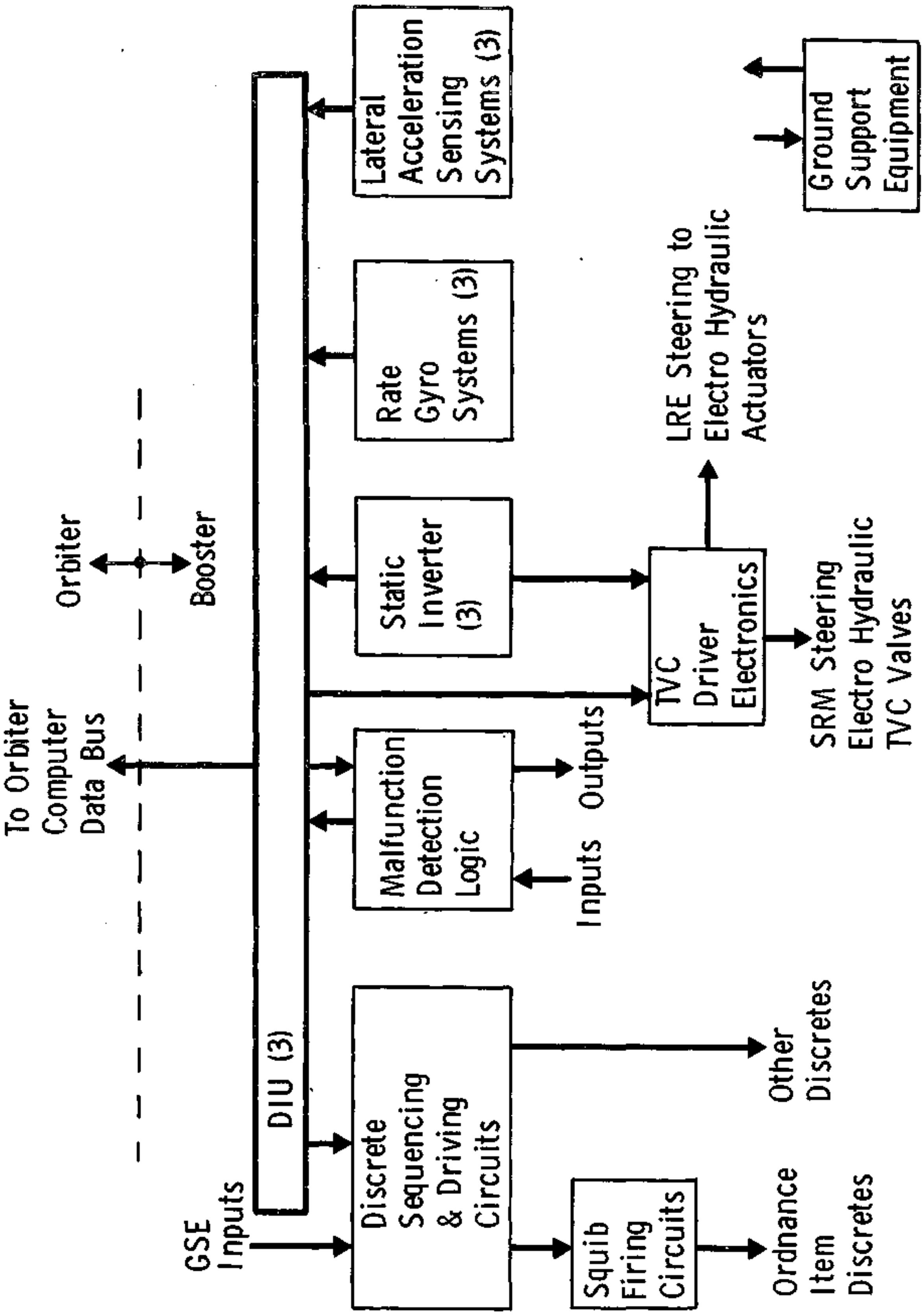


Figure IV-12 DIU/Orbiter/GSE Interfaces

b. Titan III L Instrumentation System - The instrumentation system baselined for T-III L is similar to the system used on existing Titan III vehicles. The operation and interface requirements for the instrumentation system are as follows:

1) The typical sequence of operation for propulsion measurements sensed and transmitted by the T-III L instrumentation system are shown in Figure IV-13. The measurements originate as electrical signals from sources such as pressure transducers, bus voltages or discrete switches. If the electrical signal is not compatible with the required inputs of the Remote Multiplexer Unit (RMU), it is sent to a signal conditioner where it is transformed into a compatible signal. The signal is then sampled and amplified in the RMU and, upon command from the RMIS Converter Unit (CU), it is sent to the CU in Pulse Amplitude Modulated (PAM) form. The CU transforms the PAM signal into digital form and places the digital information in a serial binary pulse train which modulates the PCM/FM transmitter. The transmitter amplifies and provides the S-Band RF carrier on which the PCM wave train is transmitted to ground receiving stations for decoding, recording and display of data.

2) Ground interfaces with the airborne instrumentation system are required for test and checkout as follows:

   a) The PCM landline outputs are used to present the RMIS outputs to the ground station without the use of the RF subsystem. This method of monitoring the PCM outputs is used during subsystem testing and when it is expedient not to radiate RF.

   b) Ground Instrumentation Equipment (GIE) must be mated directly with the RMIS converter unit to program the memory of the CU. This operation is performed to obtain the desired sample rate for all applicable measurements.

   c) Certain tank instrumentation such as temperature transducers may also be monitored during and following liquid propellant loading operations.

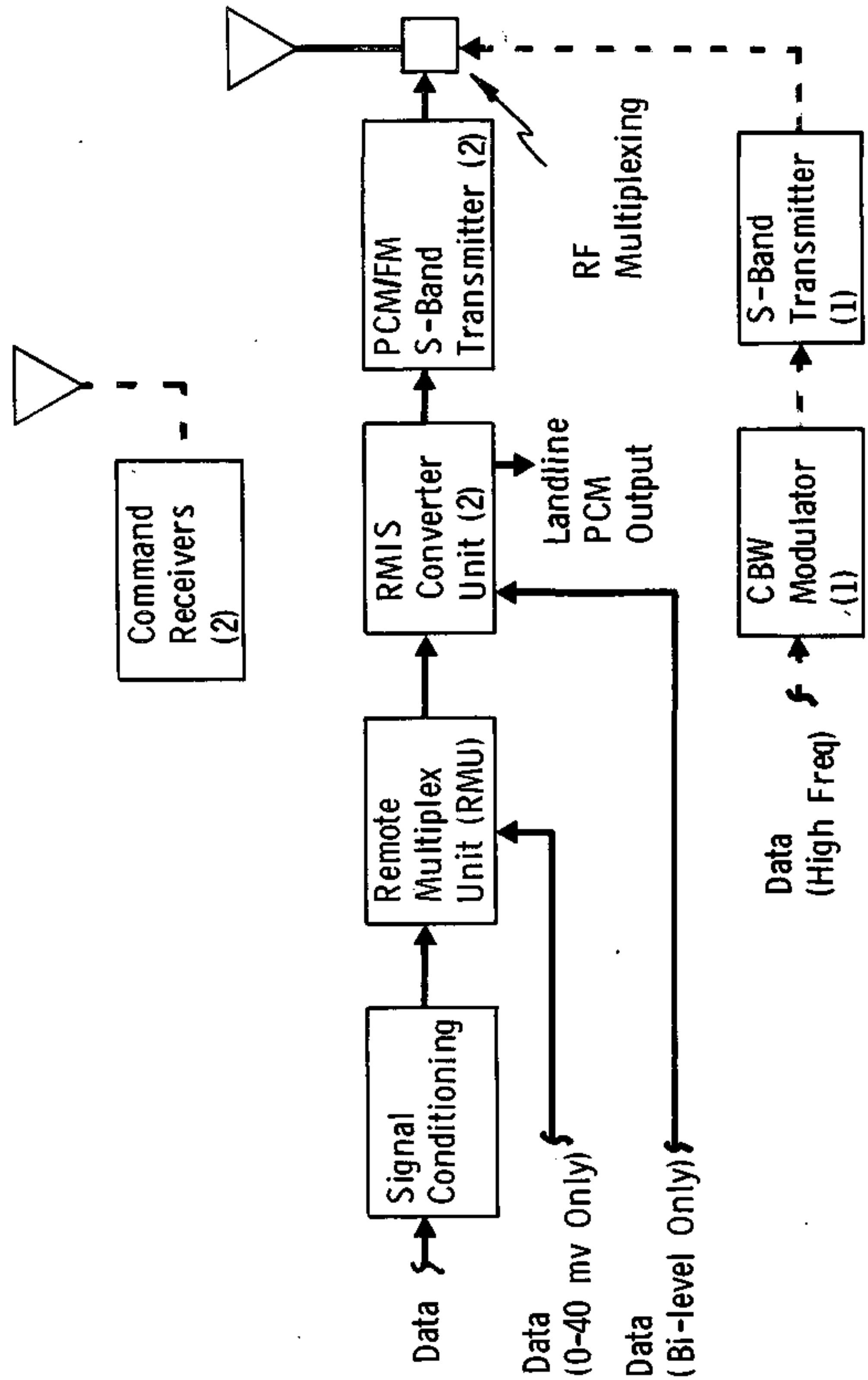


Figure IV-13 Titan III L Instrumentation

c. SRM Electrical Subsystems - The SRMs have 3 subsystems that involve avionics. The functions performed by these SRM subsystems, in conjunction with the electrical/electronic signals and components involved, are shown in Figure IV-14 and are discussed in the following paragraphs.

1) The SRM ordnance electrical subsystem provides for SRM ignition, staging, and thrust termination as well as an Inadvertent Separation Detection System (ISDS). The ignition and staging functions are performed upon receipt of discrete commands from the core vehicle. The thrust termination function is capable of being activated either by the core or by the ISDS. The ISDS provides the required logic, power and ordnance for thrust termination of the SRM should it become inadvertently separated from the core. Power for all other ordnance subsystem functions is supplied from the core Transient Power System (TPS).

2) The SRM Thrust Vector Control (TVC) subsystem is used to provide SRM steering by the injection of nitrogen tetroxide into the nozzle exit cone to deflect the exhaust gases. Twenty-four electromechanical injectant valves are arranged in groups of six around the exit cone. The valves control the flow of injectant in accordance with 0 to 10 volt command signals received from the TVC driver electronics in the core vehicle. While steering is accomplished via the injection of fluid in the individual quadrants, excess injectant fluid is dumped equally in all quadrants. TVC power is derived from the TVC battery during flight. Before launch, ground power is supplied through the ground power umbilical. On command from the ground, a motor actuated switch connects the TVC battery(s) to the power distribution bus.

3) The SRM instrumentation subsystem monitors performance of the SRM system during countdown, launch and flight and provides data for major malfunction analysis. Performance parameters, in the form of analog signals, obtained from transducers and monitors located throughout the SRM, are routed to the aft instrumentation box, and on to a remote multiplex unit. The analog signals are sampled and amplified in the RMU and upon command from the core RMIS Converter Unit (CU), the signal is sent through the forward staging disconnects to the CU in PAM form.

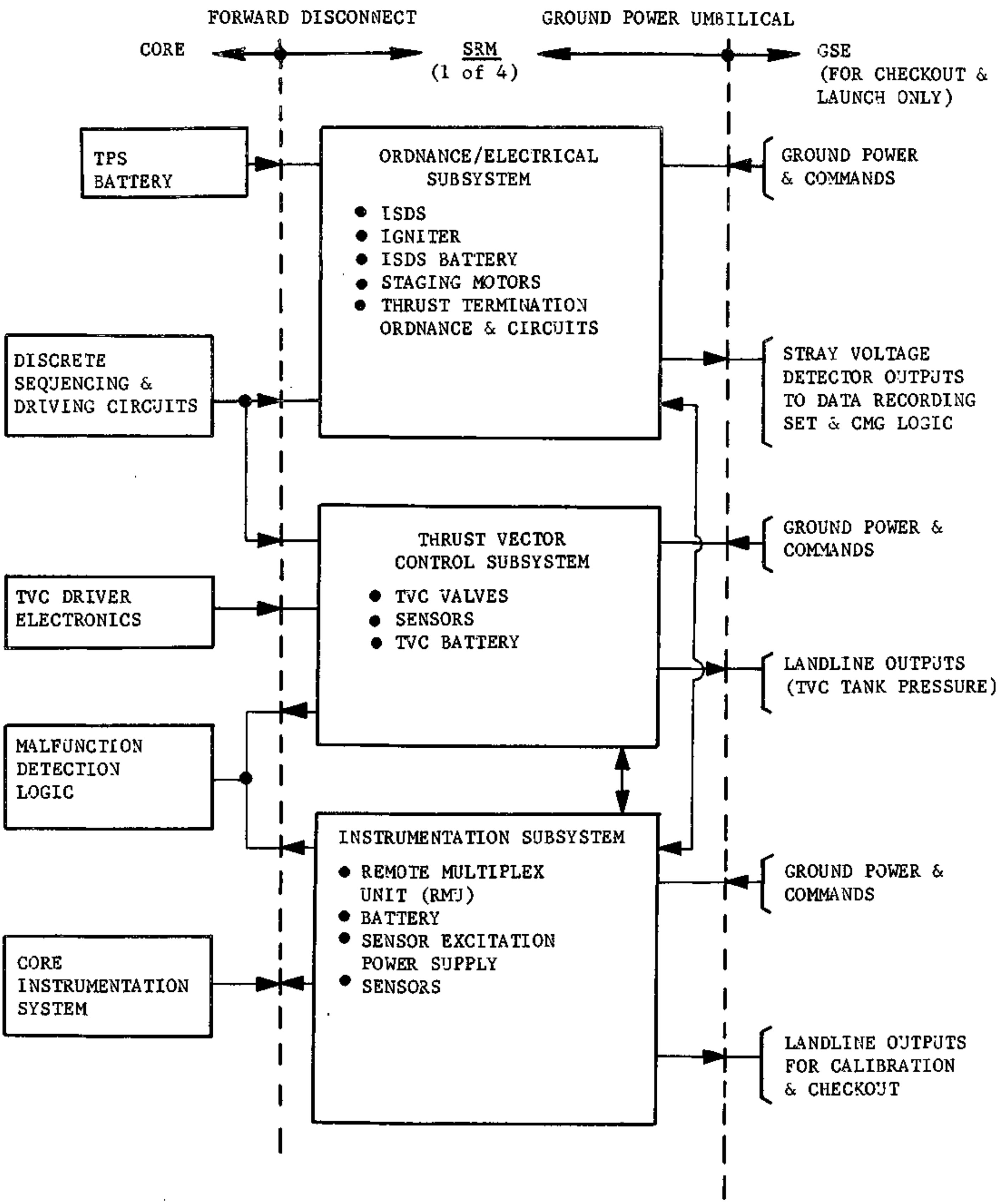


Figure IV-14 SRM Avionics

4) The SRM ground power umbilical is used to provide ground power, vehicle monitor power and commands from ground equipment. Monitors required to determine the status and performance of the ordnance electrical system for ground checkout, combined systems test, and launch countdown and holds are provided to ground facilities via the SRM ground power umbilical.

C. CHECKOUT AND MONITORING REQUIREMENTS

The results of the studies that were conducted to establish the Titan III-L propulsion checkout and monitoring requirements are presented in this section. The effort included defining the sequence and logic of propulsion system control, and performing a failure modes and effects analysis on the propulsion systems. The resultant information was then evaluated to establish the propulsion system data acquisition requirements. Also, tradeoff studies were conducted to establish criteria for allocating the functions of checkout, monitoring, and control of the propulsion systems to orbiter and booster onboard equipment and ground support equipment.

1. Propulsion Control Sequence and Logic: The propulsion control sequence and logic is illustrated in Figure IV-15. The information presented in the figure deals with the period beginning with automatic sequence terminal count (T-35 sec.) and continuing through booster shutdown and staging. A detailed discussion of the propulsion systems' functional operations was presented in Section IV-B paragraph 5.

2. Failure Modes and Effects Analysis: The ground rules used in conducting the propulsion failure modes and effects analysis (FMEA) are presented in Appendix A of this volume, together with the FMEA worksheets. In addition to defining candidate parameters for the checkout and monitoring function, the FMEA identified thirty Criticality 1 failure modes. However, the probabilities of occurence of these Criticality 1 failures are quite low, and none have been experienced on Titan III flights.

3. Data Acquisition Requirements: Candidate parameters were defined by evaluating the control sequence and logic summary and the FMEAs. Also, capability for performing a postflight analysis of propulsion system performance was considered. The resultant parameters are identified in Table IV-8. The parameters and the associated propulsion elements are listed, with the propulsion element nomenclature as defined in the propulsion configuration definition, Section IV-B. The expected range, allowable error, use of the data, and time of data activity are defined for each parameter. Response rates and sample rates are also defined and are based on current Titan III usage and capabilities.

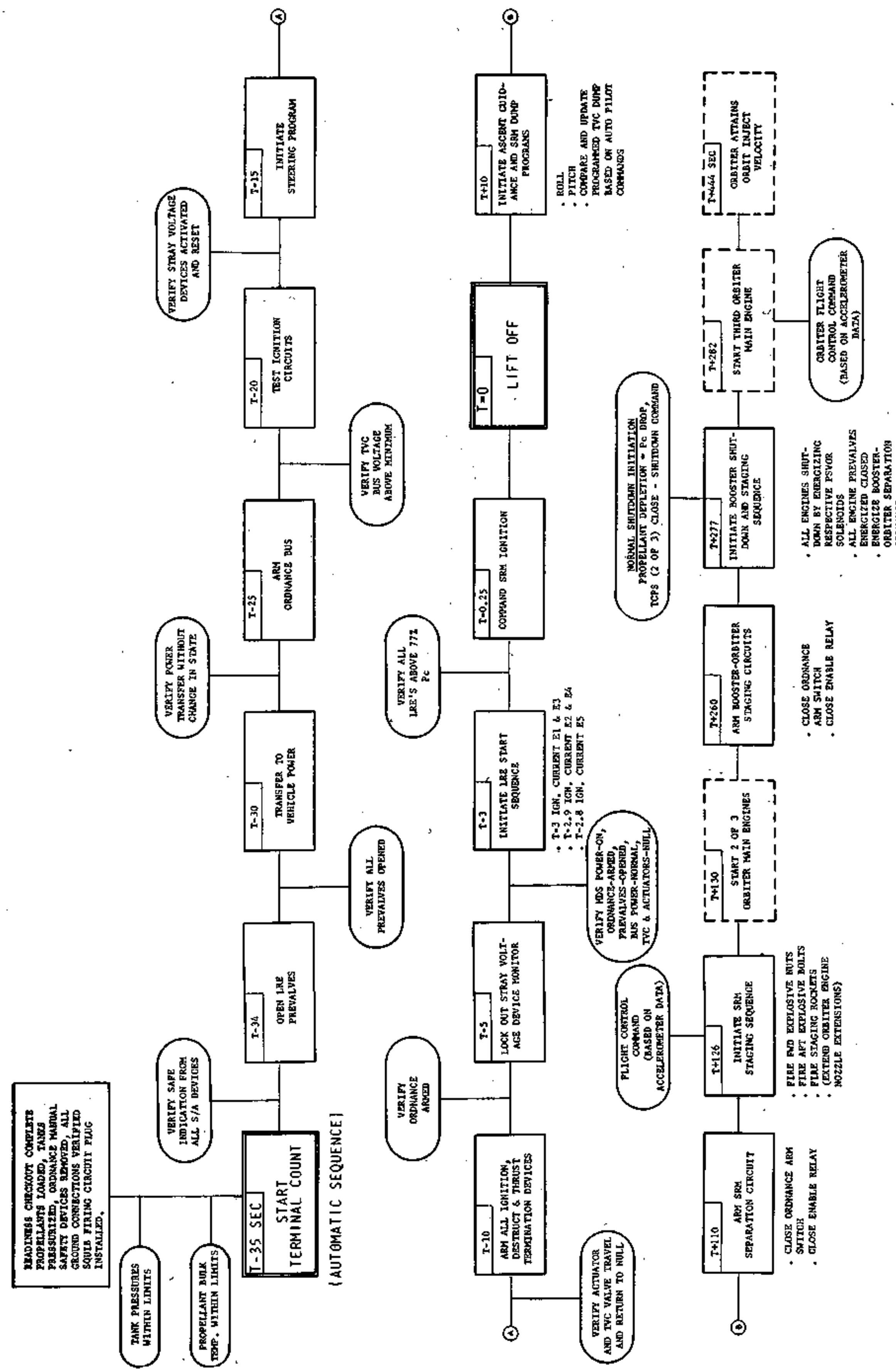


FIGURE IV-15 CONTROL SEQUENCE AND LOGIC

TABLE IV-8

DATA ACQUISITION REQUIREMENTS

PARAMETER	PROPULSION ELEMENT	PARAMETER RANGE AND UNITS	ALLOWABLE ERROR	RESPONSE RATE	SAMPLE RATE	DATA USAGE	TIME OF DATA ACTIVITY
Pressure, Oxidizer Suction, PoS	Inlet to main engine oxidizer pump (1.1.3.1)	0-200 psia 90 psia, nominal	± 6.8 psia	80 Hz	400 sps	Post flight analysis; verification of engine, oxidizor inlet pressure level and stability.	T-35 sec thru entire engine operation.
Pressure, Fuel Suction, PfS	Inlet to main engine fuel pump (1.1.3.2)	0-150 psia 40 psia, nominal	± 5.1 psia	80 Hz	400 sps	Post flight analysis; verification of engine, fuel inlet pressure level and stability.	T-35 sec thru entire engine operation.
Pressure, Thrust Chamber, P_C	Side of main engine injector (1.1.1.1)	0-1000 psia 825 psia, nominal	± 34 psia	80 Hz	400 sps	Post flight analysis; evaluation of engine performance and chamber pressure characteristics.	Total engine operating period.
Pressure, Oxidizer Discharge, PoD	Discharge side of main engine oxidizer pump (1.1.3.1)	0-1500 psia 1150 psia, nominal	± 51 psia	80 Hz	400 sps	Post flight analysis; TPA performance evaluation.	Total engine operating period.
Pressure, Fuel Discharge, PfD	Discharge of main engine fuel pump (1.1.3.2)	0-1500 psia 1350 psia, nominal	± 51 psia	80 Hz	400 sps	Post flight analysis; TPA performance evaluation.	Total engine operating period.
Pressure, Oxidizer Bootstrap Venturi Inlet, PoBTVI	Inlet to main engine, gas generator, oxidizer cavitating venturi (1.1.4.4)	0-1500 psia 1000 psia, nominal	± 51 psia	20 Hz	100 sps	Post flight analysis; gas generator performance.	Total engine operating period.
Pressure, Fuel Bootstrap Venturi Inlet, PfBTVI	Inlet to main engine, gas generator, fuel cavitating venturi (1.1.4.5)	0-1500 psia 1250 psia, nominal	± 51 psia	20 Hz	100 sps	Post flight analysis; gas generator performance.	Total engine operating period.
Pressure, Gas Generator Chamber PcGG	Side of main engine gas generator (1.1.4.1)	0-100 psia 600 psia, nominal	± 34 psia	40 Hz	200 sps	Post flight analysis; gas generator performance.	Total engine operating period.
Pressure, Fuel Pressurant Orifice Inlet, PfPOI	Inlet to main pressurization, fuel pressurization sonic nozzle (1.3.2.2)	0-500 psia 200 psia, nominal	± 17 psia	40 Hz	200 sps	Post flight analysis; performance verification of fuel autogenous pressurization assembly.	Total engine operating time.

TABLE IV-8 (CONTINUED)

DATA ACQUISITION REQUIREMENTS

PARAMETER	PROPULSION ELEMENT	PARAMETER RANGE AND UNITS	ALLOWABLE ERROR	RESPONSE RATE	SAMPLE RATE	DATA USAGE	TIME OF DATA ACTIVITY
Pressure, Oxidizer Pressurant Orifice Inlet, PoPOI	Inlet to main pressurization, oxidizer pressurization back pressure orifice (1.3.1.3)	0-1000 psia 275 psia, nominal	± 34 psia	40 Hz	200 sps	Post flight analysis; performance verification of oxidizer autogenous pressurization assembly.	Total engine operating time
Pressure, Lube Pump Discharge, PLD	Discharge of main engine lube oil pump (1.1.3.4)	0-100 psia 38 psia, nominal	± 38 psia	40 Hz	200 sps	Post flight analysis; gear box & lubricating hardware evaluation .	Total engine operating time.
Pressure, Hydraulic System, PHS	Discharge of main engine hydraulic pump (1.1.6.3)	0-4500 psia	± 210	20 Hz	100 sps	Terminal count; verification of proper operating pressure. Post flight analysis; operational integrity of TVC assembly.	T-35 sec thru total engine operating period.
Differential Pressure Pitch Actuator, PPA-D	Both sides of main engine gimbal actuator (1.1.6.2)P	-7000/+7000 psid	± 300 psid	40 Hz	200 sps	Post flight analysis; evaluation of TVC operation and contingency torques.	T-35 sec thru LRE operation.
Differential Pressure Yaw Actuator PYA-D	Both sides of main engine gimbal actuator (1.1.6.2)Y	-7000/+7000 psid	± 300 psid	40 Hz	200 sps	Post flight analysis; evaluation of TVC operation and contingency torques.	T-35 sec thru LRE operation.
Temperature, Oxidizer Suction, ToS	Inlet to main engine oxidizer pump (1.1.3.1)	0-100°F 70°F, nominal	± 5.5°F	4 Hz	20 sps	Post flight analysis; engine performance calculations.	T-35 sec thru LRE operation.
Temperature, Fuel Suction, TfS	Inlet to main engine fuel pump (1.1.3.2)	0-100°F 70°F, nominal	± 5.5°F	4 Hz	2 sps	Post flight analysis; engine performance calculations.	T-35 sec thru LRE operation.
Temperature, Fuel Pressurant Orifice Inlet, TfPOI	Inlet to main engine fuel pressurization sonic nozzle (1.3.2.2)	0-500°F 240°F, nominal	± 17°F	4 Hz	20 sps	Post flight analysis; performance of fuel autogenous pressurization.	Total engine operation.
Temperature, Oxidizer Pressurant Orifice Inlet, ToPOI	Inlet to main engine oxidizer pressurization back pressure orifice (1.3.1.3)	0-500°F 250°F, nominal	± 17°F	4 Hz	20 sps	Post flight analysis; performance of oxidizer autogenous pressurization.	Total engine operation.

TABLE IV-8 (CONTINUED)

DATA ACQUISITION REQUIREMENTS

PARAMETER	PROPULSION ELEMENT	PARAMETER RANGE AND UNITS	ALLOWABLE ERROR	RESPONSE RATE	SAMPLE RATE	DATA USAGE	TIME OF DATA ACTIVITY
Temperature, Nozzle Extension Skin, TNES-1, 3, 5	Main engine ablative nozzle (1.1.1.3)	0-1000°F 190°F, nominal	± 34°F	4 Hz	20 sps	Post flight analysis; integrity verification of ablative nozzle and refrasil insulation.	Total engine operation.
Temperature, Gear Box Bearing, No. 6A, 6B TGB-6A, B	Main engine turbopump assembly gear box (1.1.3.3)	0-500°F 200°F, nominal	± 17°F	40 Hz	200 sps	Post flight analysis; caution and warning display to crew.	Total engine operating period.
Thrust Chamber Pressure Switch, TCPS-A, B, C	Main engine injector dome (1.1.1.1)	Increasing pressure (620 ± 40 psia) Decreasing pressure (600 ± 60 psia)	± 40 psia ± 60 psia	$<.\overline{01}$ sec $<.\overline{01}$ sec	Discrete (100 sps) Discrete (100 sps)	Used for engine start and shutdown logic, liftoff and staging logic and for C&W and fault isolation logic.	Total LRE operating period.
Speed, Turbine, NT	Main engine turbopump assembly gear box (1.1.3.3)	0-31000 RPM (23,000 RPM, nominal)	± 1054 RPM	8 Hz	40 sps	Post flight analysis; turbo pump assembly performance evaluation	Total LRE operating period.
Level, Hydraulic Reservoir, LHS	Main engine hydraulic reservoir above hydraulic pump (1.1.6.3)	0-100%	3.4%	4 Hz	20 sps	Preflight and flight verification of hydraulic system.	T-35 sec through LRE operation.
Position, Oxidizer Prevalve, LoPV	Main propellant management oxidizer prevalve (1.2.1.3)	Open/Closed	-	-	Discrete	Indication to crew and ground personnel of valve position.	Propellant loading; T-35 sec to core separation.
Position, Fuel Prevalve, LfPV	Main propellant management fuel prevalve (1.2.2.3)	Open/Closed	-	-	Discrete	Indication to crew and ground personnel of valve position.	T-35 sec to core separation.
Position, Pitch Actuator, LPA	Main engine gimbal actuator (1.1.6.2)P	± 5 degrees	0.2 degrees	20 Hz	100 sps	Verify engine in proper position for start and compare position to commands.	T-35 sec to LRE shutdown.
Position, Yaw Actuator, LYA	Main engine gimbal actuator (1.1.6.2)Y	± 5 degrees	0.2 degrees	20 Hz	100 sps	Verify engine in proper position for start and for comparing position to commands.	T-35 sec to LRE shutdown.
Detector, Engine Compartment Fire/ Leakage DECF/DgEC	Engine compartment at each main engine (1.1)	On/Off	-	-	Discrete	Alert crew hazardous condition within the LRE compartment. NOTE: This detector is idenfified as Research & Technology item requiring further eval.	T-35 sec to core separation.

TABLE IV-8 (CONTINUED)

DATA ACQUISITION REQUIREMENTS

PARAMETER	PROPULSION ELEMENT	PARAMETER RANGE AND UNITS	ALLOWABLE ERROR	RESPONSE RATE	SAMPLE RATE	DATA USAGE	TIME OF DATA ACTIVITY
Pressure, Oxidizer Tank, Gas, PgOT-1, 2	Main propellant management oxidizer tank (1.2.1.1), Top.	0-50 psia 35 psia, nominal	± 1.7 psia	8 Hz	40 sps	Verify tank pressure is sufficient for structural integrity and LRE operation. Caution and warning indication to crew.	From tank pre-pressurization to LRE shut-down.
Pressure, Fuel Tank, Gas, PgFT-1, 2	Main propellant management fuel tank (1.2.2.1), Top.	0-50 psia 27 psia, nominal	± 1.7 psia	8 Hz	40 sps	Verify tank pressure is sufficient for structural integrity and LRE operation. Caution and warning indication to crew.	From tank pre-pressurization to LRE shut-down.
Temperature, Oxidizer Tank, Liquid, ToT	Main propellant management oxidizer tank (1.2.1.1)	0-150°F 110°F, nominal	± 1.7°F	4 Hz	20 sps	Post flight analysis; check of total system performance.	T-10 sec to core separation.
Temperature, Fuel Tank, Liquid, TfT	Main propellant management fuel tank (1.2.2.1)	0-150°F 70°F, nominal	± 1.7°F	-	Land line analog	Propellant bulk temperature evaluation required for launch commitment.	Propellant loading to liftoff.
Pressure, Motor Head End, $PgMHE_{A, B}$	SRM assembly forward closure (2.1.1.1)	0-1000 psia	± 34 psia	80 Hz	400 sps	Post flight analysis; for SRM performance verification. Caution and warning indication to crew.	SRM ignition to SRM staging.
Pressure, Injectant Manifold, PoIM	SRM TVC tank assembly in manifold downstream of injectant transfer tube (2.2.1.4)	0-1500 psia	± 51 psia	80 Hz	400 sps	Post flight analysis; for TVC performance verification.	SRM ignition to SRM staging.
Pressure, Injectant Tank, Gas, PgIT	SRM TVC tank assembly injectant tank (2.2.1.1)	0-1500 psia	51 psia	-	Land line	Used for tank pressure integrity verification.	From tank pressurization to liftoff.
Position, Injectant Valve, Average, Quad 1 thru 4, A & B	SRM TVC injectant valve, (2.2.2.1)	0-9.5 VDC	± 0.32 VDC	20 Hz	100 sps	Verify valve positions corresponds to steering commands; post flight analysis.	SRM burn.
Position, SRM S/A Device, L-S/A	SRM assembly rocket motor igniter (2.1.1.5) safe and arm device.	Safe/Armed	-	-	Discrete Land line	Indicates igniter safe or arm status.	T-35 sec to liftoff

TABLE IV-8 (CONTINUED)

DATA ACQUISITION REQUIREMENTS

PARAMETER	PROPULSION ELEMENT	PARAMETER RANGE AND UNITS	ALLOWABLE ERROR	RESPONSE RATE	SAMPLE RATE	DATA USAGE	TIME OF DATA ACTIVITY
Detector, SRM Burnthrough, DBT	Rocket motor subsystem SRM assembly (2.1.1)	On/Off	-	-	Discrete	Caution and warning indication - possible abort initiate. <u>NOTE</u>: This detector is identified as a supporting research and technology item requiring further evaluation.	Total SRM burn.

NOTE: The selected sample and response rates are derived from current Titan III sensor usage and capabilities

4. Function Incorporation: Evaluations were conducted to define the recommended approach and supporting rationale for performing the functions of control, checkout and monitoring of the Titan III-L propulsion systems.

   a. Propulsion System Control: Table IV-9 identifies propulsion system control functions, applicable candidates for initiating the control functions, and the recommended approach for each. The control functions are categorized in the table according to mission phase, i.e., prior to mating of the booster and orbiter, postmate/preflight, and inflight. It is recommended that ground equipment be used to initiate the control functions during ground operations up to start of automatic sequence terminal count. The rationale supporting this judgment is that a proven design for ground equipment and basic operating procedures exist for accommodating the control functions associated with subsystem and system checkout, propellant loading and pressurization. It is further recommended that ground equipment be used for the automatic sequence terminal count and launch execution control functions for the same reason. However, these control functions can be performed from the orbiter if launch operations requirements so dictate.

      The inflight propulsion control functions are allocated as shown in the table. Thrust vector control and SRM separation will be initiated from the vehicle flight control system, which is located in the orbiter. SRM thrust termination for boost phase abort will also be initiated from the orbiter, since this function must originate from a flight-crew decision or from central computer complex logic. SRM destruct in case of inadvertant SRM separation will originate from the SRM's inadvertant separation destruct system (ISDS). Normal liquid engine shutdown will originate from the booster in the case of propellant depletion, but can originate in the orbiter when the required boost phase velocity is achieved. Individual booster engine shutdown can be initiated by the booster (low chamber pressure) or from the orbiter. Range safety booster destruct is a ground-originated function.

TABLE IV-9

CONTROL FUNCTION IMPLEMENTATION

TIME PHASE AND OPERATION	CONTROL CANDIDATES	RECOMMENDATION
Prior to Booster Orbiter Mate		
1. Functional and Leak Tests	Ground, Booster	Ground
Post Mate/Preflight		
1. Combined Systems Test	Ground, Orbiter	Ground
2. Propellant Loading & Pressurization	Ground, Orbiter	Ground
3. Terminal Count and Launch Execution	Ground, Orbiter	Ground (or Orbiter)
a. Prevalve Actuation, Engine & SRM Start		
Inflight		
1. TVC	Orbiter	Orbiter
2. SRM Thrust Termination (Boost Phase Abort)	Orbiter, Booster	Orbiter
3. Inadvertant SRM Separation - SRM Destruct	Orbiter, Booster	Booster (SRM)
4. SRM & Core Destruct (Range Safety)	Ground, Orbiter	Ground
5. SRM Separation	Orbiter, Booster	Orbiter
6. Engine Shutdown (Normal)	Orbiter, Booster	Booster or Orbiter
7. Individual Engine Shutdown (Fault)	Orbiter, Booster	Booster or Orbiter

b. Propulsion Systems Checkout: The checkout function is performed during ground operations to assure flight readiness (there is no inflight checkout of the Titan III-L propulsion systems, since operation of the propulsion system is initiated prior to lift off). The booster propulsion systems checkout can be accomplished by using conventional Titan III ground support equipment (GSE), by incorporating the checkout function into the orbiter, or by developing and using new booster GSE. (Incorporating the checkout function on board the booster is not an option in this case, since the booster is unmanned and non-recoverable.) Table IV-10 delineates certain advantages and disadvantages for each of the three options. It is recommended that conventional Titan III GSE be employed for ground checkout of the booster propulsion systems. The driving factor for this recommendation is the low acquisition cost and the degree of cost credibility of this approach; this factor is commensurate with the use of the Titan III-L expendable booster for Space Shuttle.

c. Propulsion Systems Monitoring: The monitoring function is performed during ground operations to assure that designed parameters (such as tank pressures) are within their specified limits, and during flight for fault detection and for performance data acquisition. During ground operations (after booster/orbiter mating), the booster propulsion system monitoring function can be performed by ground support equipment (GSE), by orbiter onboard equipment, or by a combination of these two. Similarly, during flight the monitoring can be accomplished with telemetry, by orbiter onboard equipment, or by a combination.

In correspondence to the recommendations to employ ground support equipment for the ground control and checkout functions, it is recommended that such equipment also be used for the ground monitoring function. This conclusion is again based on the logic that the basic design and basic operating procedures of the equipment has previously been developed and used on the Titan III program.

From the information presented in Table IV-8, Data Acquisition Requirements, it can be seen that nearly all of the inflight propulsion monitoring data is to be used for propulsion performance analysis on a postflight basis.

TABLE IV-10

PROPULSION SYSTEMS CHECKOUT

ADVANTAGES	DISADVANTAGES
<u>Conventional Titan III GSE</u>	
. Operational capability established. . Credible cost evaluation. . Minimum airborne software. . Technically fulfills requirements. . Established processes and operational procedures. . Will support booster systems tests. . Would minimize orbiter weight and development cost.	. Small design change to incorporate new parts. . Must be rated for manned vehicle support. . Not inherently flexible to support other systems. . Offers no advance in technology. . Cannot accommodate the inflight propulsion monitoring function.
<u>Orbiter Onboard Equipment</u>	
. Minimum ground systems hardware and operations cost. . Maximum system adaptability to program evolutions. . Maximum integration of orbiter/booster systems. . Can incorporate the inflight monitoring function.	. Maximum airborne software concept. . Two contractors involved to accomplish booster task. . All new checkout procedures.
<u>New Booster GSE</u>	
. Advanced technology possible. . Has evolution possibilities for booster changes. . Can support booster systems test.	. Cannot incorporate the inflight monitoring function. . New procedures, crew training, and software. . Higher cost and lower cost credibility than conventional Titan GSE.

Telemetry is used for this purpose on Titan III vehicles; the Titan III design could be adapted to the Titan III L configuration through expansion to accommodate the larger number of engines and SRMs. This is the recommended approach, with the remaining data (to be acquired for fault detection and crew caution and warning) accommodated by orbiter onboard monitoring.

D. CHECKOUT AND MONITORING IMPLEMENTATION

The following paragraphs delineate the selected Titan III L propulsion measurements and sensors, and describe the checkout approach and equipment.

1. Measurements and Sensors: Table IV-12 presents selected measurements, identifies the source from which the measurement requirement was derived, and summarizes the justification for the measurement. Table IV-13 defines the usage of the data. Table IV-14 describes the corresponding sensors. The locations of the measurements are shown schematically in Figure IV-16 for the main propulsion system, and in Figure IV-17 for the Solid Rocket Motors. Measurement quantities are summarized in Table IV-11.

TABLE IV-11

MEASUREMENT QUANTITIES

	CONTROL	FAULT DETECTION, CAUTION AND WARNING	PERFORMANCE ANALYSIS	TOTAL
Liquid Engine (each)	1	3	24	28
Main Propulsion	15	20	113	148
Solid Rocket Motor (each)	5	2	3	10
Propulsion System Total	35	28	125	188

NOTE: Redundancies not included.

C.3

TABLE IV-12 Measurement Selection Criteria

MEASUREMENT	REQUIREMENT SOURCE	USAGE JUSTIFICATION
Pressure, Oxidizer Suction, PoS	FMEA - 1.3.1.7; Checkout and monitoring requirements analysis, parameter optimization study.	Pump inlet pressures are required for post flight analysis of system hydraulic oscillations and propellant depletion determination. The measurement can also be used to verify tank pressures.
Pressure, Fuel Suction, PfS	FMEA - 1.3.1.7; Checkout and monitoring requirements analysis, parameter optimization study.	Same as above.
Pressure, Thrust Chamber, P_C	FMEA - 1.1.2.1 (Fuel), 1.1.2.2 (OX), 1.1.2.3, 1.1.3.1, 1.1.3.2, 1.1.3.3, 1.1.4.1, 1.1.4.2, 1.1.4.3, 1.1.4.4, 1.1.4.5, 1.1.4.6, 1.1.4.7, 1.1.5.1, 1.1.5.2; Checkout and monitoring requirements analysis.	Thrust chamber pressure is required for calculating engine performance parameters. This measurement is also used to detect various modes of engine and engine component failures.
Pressure, Oxidizer Discharge	FMEA - 1.1.3.1, 1.1.3.3, 1.1.3.6; Checkout and monitoring requirements analysis.	Measurement is used to determine pump performance, to calculate flowrate and to detect failures in the pump or related hardware such as the gear box.
Pressure, Fuel Discharge, PfD	FMEA - 1.1.3.2, 1.1.3.3, 1.1.3.6; Checkout and monitoring requirements analysis.	Same as above.
Pressure, Oxidizer Bootstrap Venturi Inlet, P_oBTVI	FMEA - 1.1.4.4; Checkout and monitoring requirements analysis, parameter optimization study.	Measurement is needed to analyze gas generator loop performance.
Pressure, Fuel Bootstrap Venturi Inlet, PfBTVI	FMEA - 1.1.4.4; Checkout and monitoring requirements analysis, parameter optimization study.	Same as above.
Pressure, Gas Generator Chamber PcGG	FMEA - 1.1.4.1, 1.1.4.4, 1.1.4.5, 1.1.4.6, 1.1.4.7; Checkout and monitoring requirements analysis.	This pressure measurement is required for establishing the gas generator performance and to analyze associated problem areas.

TABLE IV-12 (CONTINUED)

MEASUREMENT	REQUIREMENT SOURCE	USAGE JUSTIFICATION
Pressure, Fuel Pressurant Orifice Inlet, PfPOI	FMEA - 1.3.2.1, 1.3.2.2, 1.3.2.4; Checkout and monitoring requirements analysis.	Information from this measurement is required for acceptance and postflight analysis to calculate autogenous flowrate. Data is also used to verify engine burst disc rupture and system integrity after engine shutdown.
Pressure, Oxidizer Pressurant Orifice Inlet, PoPOI	FMEA - 1.3.1.3, 1.3.1.5; Checkout and monitoring requirements Analysis.	Same as above.
Pressure, Lube Pump Discharge	FMEA - 1.1.3.5; Checkout and monitoring requirements analysis, parameter optimization analysis.	This measurement provides correlating information for condition of the gear box.
Pressure, Hydraulic System, PHS	FMEA - 1.1.6.1, 1.1.6.4; Checkout and monitoring requirements analysis.	Hydraulic pressure data is required to verify subsystem integrity for launch and flight operation.
Differential Pressure YAW Actuator, PYA-D	FMEA - 1.1.6.1, 1.1.6.2, 1.1.6.4; Checkout and monitoring requirements analysis.	Same as above.
Temperature, Oxidizer Suction, ToS	Parameter optimization analysis	Required for postflight analysis for propellant density determination.
Temperature, Fuel Suction, TfS	Same as above.	Same as above.
Temperature, Fuel Pressurant Orifice Inlet, TFPOI	FMEA - 1.3.2.1, 1.3.2.2; Checkout and monitoring requirements analysis.	Data is required for postflight analysis to determine autogenous gas temperature and gas cooler performance.
Temperature, Oxidizer Pressurant Orifice Inlet, TOPOI	FMEA - 1.3.1.3; Checkout and monitoring requirements analysis.	Data is required for postflight analysis to determine autogenous gas energy level and superheater performance.

TABLE IV-12 (CONTINUED)

MEASUREMENT	REQUIREMENT SOURCE	USAGE JUSTIFICATION
Temperature, Nozzle Extension Skin TNES - 1, 3, 5	FMEA - 1.1.1.2; Checkout and monitoring requirements analysis.	This measurement is required for determining the nozzle insulation performance and integrity of the nozzle extension.
Temperature, Gear Box Bearing No. 6A, 6B TGB - 6A, B	FMEA - 1.1.3.5; Checkout and monitoring performance analysis.	This measurement is required for indication of impending gear box failure and bearing trend analysis (used for caution & warning).
Thrust Chamber Pressure Switch TCPS - A, B, C	FMEA - 1.1.1.4, 1.1.2.1 (Fuel), 1.1.2.2 (Ox), 1.1.2.3, 1.1.3.6, 1.1.4.2, 1.1.4.3, 1.1.5.1, 1.1.5.2; Checkout and monitoring requirements analysis, control sequence and logic study.	Switches are required for engine start and shutdown verification, liftoff and staging sequencing, fault detection, caution and warning, and fault isolation.
Speed, Turbine, NT	FMEA - 1.1.3.3, 1.1.3.5, 1.1.3.6, 1.1.4.1; Checkout and monitoring requirements analysis.	Measurement is required for postflight evaluation of the gear box and gas generator loop and TVC analysis.
Level, Hydraulic Reservoir, LHS	Operational flow analysis, parameter optimization study.	This measurement is required to verify flight control system readiness and inflight operation.
Position, Oxidizer Prevalve, LOPV	FMEA - 1.2.1.3, 1.2.2.3; Control sequence and logic analysis.	Required for verification of prevalve position for propellant loading, prior to engine ignition, and engine isolation after shutdown.
Position, Fuel Prevalve, LFPV	FMEA - 1.2.1.3, 1.2.2.3; Control sequence and logic analysis.	Same as above.
Position, Pitch Actuator, LPA	FMEA - 1.1.6.1, 1.1.6.2; Checkout and monitoring requirements analysis, parameter optimization study.	Measurement required for engine position indication for start and evaluation of steering command response.

TABLE IV-12 (CONTINUED)

MEASUREMENT	REQUIREMENT SOURCE	USAGE JUSTIFICATION
Position, Yaw Actuator, LYA	FMEA - 1.1.6.1, 1.1.6.2; Checkout and monitoring requirements analysis, parameter optimization study.	Same as above.
Detector, Engine Compartment Fire/ Leakage, DECF/DgEC	FMEA - 1.1.3.2, 1.1.3.4, 1.1.4.1, 1.1.4.5, 1.1.5.2, 1.2.2.3, 1.3.2.1; Checkout and monitoring requirements analysis.	Required for fire/leakage detection in the engine compartment. Provide input to caution and warning.
Pressure, Oxidizer Tank, Gas, PgOT-1, 2	FMEA - 1.1.4.1, 1.3.2.1, 1.3.2.3, 1.3.2.4, 1.3.2.5, 1.3.2.6, 1.3.1.1, 1.3.1.2, 1.3.1.3, 1.3.1.4, 1.3.1.5, 1.3.1.7; Checkout and monitoring requirements analysis, control sequence and logic analysis.	Needed to verify status of tank gas pressure for structural integrity and engine inlet conditions. Data inputs utilized for crew caution and warning and range safety.
Pressure, Fuel Tank Gas, PgFT - 1, 2	Same as above.	Same as above.
Temperature, Oxidizer Tank, Liquid, TfT	FMEA - 1.3.2.1; Control sequence and logic analysis.	Required for launch commitment, i.e., propellant bulk temperature limit verification.
Temperature, Fuel Tank, Liquid, TfT	Control logic and sequence analysis.	Same as above.
Pressure, Motor Head End, $PgMHE_{A, B}$	FMEA - 2.1.1.1, 2.1.1.2, 2.1.1.3, 2.1.1.6; Checkout and monitoring requirements analysis.	Required for SRM performance verification, monitoring and malfunction detection.
Pressure, Injectant Manifold, PoIM	Checkout and monitoring requirements analysis (FMEA - 2.1.1.4, 2.2.1.2).	Required for TVC performance verification and future N_2O_4 dump program adjustments.
Pressure, Injectant Tank, Gas, PgIT	Checkout and monitoring requirements analysis, control sequence and logic analysis.	Required for injectant tank integrity verification and determination of proper tank pressurization prior to launch.

TABLE IV-12 (CONTINUED)

MEASUREMENT	REQUIREMENT SOURCE	USAGE JUSTIFICATION
Position, Injectant Valve, Average, Quad 1 thru 4, A & B	FMEA - 2.2.2.1; Control sequence and logic analysis.	Needed to verify response to steering commands and fault isolation of malfunctioning valve.
Position, SRM S/A Device, L-S/A	FMEA - 2.1.1.5; Safety requirements study, control sequence and logic analysis.	Required for S/A position determination for safety and operational purposes.
Detector, SRM Burnthrough, DBT	FMEA - 2.1.1.1, 2.1.1.2, 2.1.1.3, 2.1.1.4; Checkout and monitoring requirements analysis.	Needed to detect hazardous condition for caution and warning information.

TABLE IV-13

MEASUREMENT USAGE

CORE LRE MEASUREMENTS	SYMBOL	Telemetry Inputs	GSE INPUTS: Sequence/ Control Logic	GSE INPUTS: Landline In-strumentation	ORBITER INPUTS: Fault Detection Logic	ORBITER INPUTS: Display	ORBITER INPUTS: Sequence Logic	Booster Malfunction Detection Logic
Pressure, Oxidizer Suction	PoS	X						
Pressure, Fuel Suction	PfS	X						
Pressure, Thrust Chamber	Pc	X						
Pressure, Oxidizer Discharge	PoD	X						
Pressure, Fuel Discharge	PfD	X						
Pressure, Oxidizer Bootstrap Venturi Inlet	PoBTVI	X						
Pressure, Fuel Bootstrap Venturi Inlet	PfBTVI	X						
Pressure, Gas Generator Chamber	PcGG	X						
Pressure, Fuel Pressurant Orifice Inlet	PFPOI	X						
Pressure, Oxidizer Pressurant Orifice Inlet	POPOI	X						
Pressure, Lube Pump Discharge	PLD	X						
Pressure, Hydraulic System	PHS	X						
Differential Pressure, Pitch Actuator	PPA-D	X						
Differential Pressure, Yaw Actuator	PYA-D	X						
Temperature, Oxidizer Suction	ToS	X						
Temperature, Fuel Suction	TfS	X						
Temperature, Fuel Pressurant Orifice Inlet	TFPOI	X						
Temperature, Oxidizer Pressurant Orifice Inlet	TOPOI	X						
Temperature, Nozzle Extension Skin-1	TNES-1	X						
Temperature, Nozzle Extension Skin-3	TNES-3	X						
Temperature, Nozzle Extension Skin-5	TNES-5	X						
Temperature, Gear Box Bearing No. 6A	TGB-6A	X			X	X		

TABLE IV-13 (CONTINUED)

MEASUREMENT USAGE

CORE LRE MEASUREMENTS (CONTINUED)	SYMBOL	TELEMETRY INPUTS	GSE Inputs: Sequence/ Control Logic	GSE Inputs: Landline Instrumentation	Orbiter Inputs: Fault Detection Logic	Orbiter Inputs: Display	Orbiter Inputs: Sequence Logic	Booster Malfunction Detection Logic
Temperature, Gear Box Bearing No. 6B	TGB-6B	X			X	X		X
Thrust Chamber Pressure Switch A	TCPS-A	X						
Thrust Chamber Pressure Switch B	TCPS-B	X	X		X	X	X	X
Thrust Chamber Pressure Switch C	TCPS-C	X						X
Speed, Turbine	NT	X						
Level, Hydraulic Reservoir	LHS	X						
Position, Oxidizer Prevalve	LOPV	X	X					
Position, Fuel Prevalve	LFPV	X	X					
Position, Pitch Actuator	LPA	X						
Position, Yaw Actuator	LYA	X						
Detector, Engine Compartment Fire/Leakage	DECF/DgEC	X			X	X		
CORE TANK MEASUREMENTS								
Pressure, Oxidizer Tank Gas-1	PgOT-1	X			X	X		
Pressure, Fuel Tank Gas-1	PgFT-1	X			X	X		
Pressure, Oxidizer Tank-2	PgOT-2	X			X	X		
Pressure, Fuel Tank Gas-2	PgFT-2	X			X	X		
Temperature, Oxidizer Tank Liquid	ToT			X				
Temperature, Fuel Tank Liquid	TfT			X				

TABLE IV-13 (CONTINUED)

MEASUREMENT USAGE

SRM MEASUREMENTS	SYMBOL	Telemetry	GSE Inputs: Sequence/ Control Logic	GSE Inputs: Landline Instrumentation	Orbiter Inputs: Fault Detection Logic	Orbiter Inputs: Display	Orbiter Inputs: Sequence Logic	Booster Malfunction Detection Logic
Pressure, Motor Heat End, A	PgMHEA	XX			X	X		
Pressure, Motor Head End, B	$PgMHE_B$	XX						
Pressure, Injectant Manifold	PoIM	X						
Pressure, Injectant Tank Gas	PgIT			X				
Position, Injectant Valve, Average Quad 1, A	$LIVA1_A$	X					X	X
Position, Injectant Valve, Average Quad 1, B	$LIVA1_B$	X					X	X
Position, Injectant Valve, Average Quad 2, A	$LIVA2_A$	X					X	X
Position, Injectant Valve, Average Quad 2, B	$LIVA2_B$	X					X	X
Position, Injectant Valve, Average Quad 3, A	$LIVA3_A$	X					X	X
Position, Injectant Valve, Average Quad 3, B	$LIVA3_B$	X					X	X
Position, Injectant Valve, Average Quad 4, A	$LIVA4_A$	X					X	X
Position, Injectant Valve, Average Quad 4 B	$LIVA4_B$	X					X	X
Position, SRM S/A Device	L-S/A	X	X					
Detector, SRM Burnthrough	DBT	X			X	X	X	

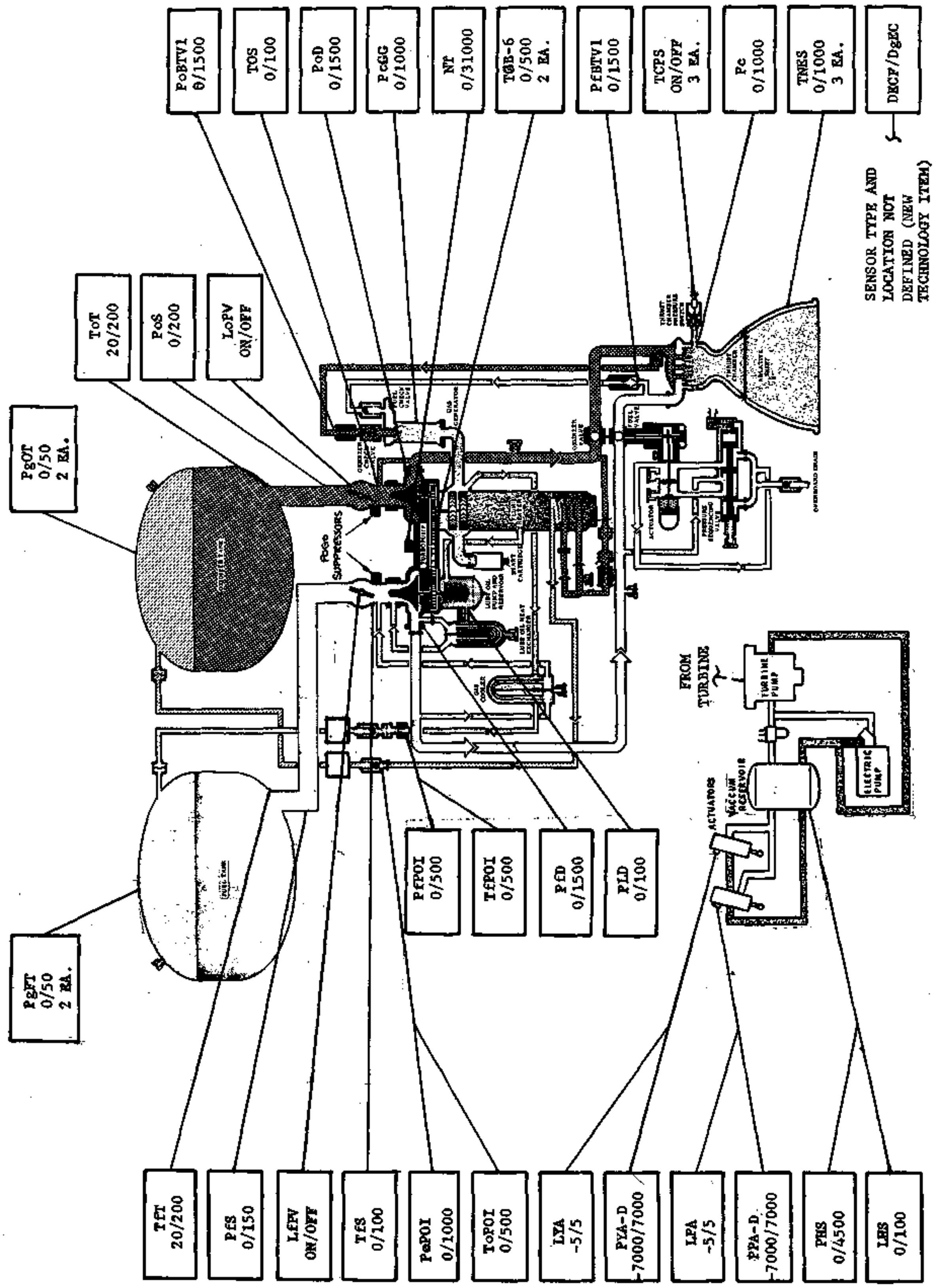
TABLE IV-14

PROPULSION SENSORS

CORE LRE MEASUREMENTS	SYMBOL	RANGE & UNITS		USED ON PREVIOUS TITAN III VEHICLES	TECHNOLOGY EXISTS	NEW TECHNOLOGY REQUIRED	TRANSDUCER TYPE	ACCURACY IN EQUIVALENT ENGINEERING UNITS (±)		A/B TLM MEAS RESPONSE		APPLICABLE NOTES
								TRANSDUCER ONLY	TOTAL INSTRUMENTATION SYSTEM	SAMPLE RATE IN SAMPLES PER SECOND	EQUIVALENT MAXIMUM RESPONSE	
Pressure, Oxidizer Suction	PoS	0/200	PSIA	NO	YES	-	Strain Gage Bridge	3.0	4.8	400	80 Hz	1, 3
Pressure, Fuel Suction	PfS	0/150	PSIA	YES	-	-	Strain Gage Bridge	2.24	3.64	400	80 Hz	1, 2
Pressure, Thrust Chamber	Pc	0/1000	PSIA	YES	-	-	Strain Gage Bridge	23.4	31.0	400	80 Hz	1, 2
Pressure, Oxidizer Discharge	PoD	0/1500	PSIA	YES	-	-	Strain Gage Bridge	35.0	43.5	400	80 Hz	1, 2
Pressure, Fuel Discharge	PfD	0/1500	PSIA	YES	-	-	Strain Gage Bridge	35.0	43.5	400	80 Hz	1, 2
Pressure, Oxidizer Bootstrap Venturi Inlet	PoBTVI	0/1500	PSIA	YES	-	-	Strain Gage Bridge	35.0	43.5	200	40 Hz	1, 2
Pressure, Fuel Bootstrap Venturi Inlet	PfBTVI	0/1500	PSIA	YES	-	-	Strain Gage Bridge	35.0	43.5	200	40 Hz	1, 2
Pressure, Gas Generator Chamber	PcGG	0/1000	PSIA	YES	-	-	Strain Gage Bridge	23.4	31.0	200	40 Hz	1, 2
Pressure, Fuel Pressurant Orifice Inlet	PfPOI	0/500	PSIA	YES	-	-	Strain Gage Bridge	12.7	16.5	100	20 Hz	1, 2
Pressure, Oxidizer Pressurant Orifice Inlet	PoPOI	0/1000	PSIA	YES	-	-	Strain Gage Bridge	23.4	31.0	100	20 Hz	1, 2
Pressure, Lube Pump Discharge	PLD	0/100	PSIA	YES	-	-	Strain Gage Bridge	.76	3.7	100	20 Hz	1, 2
Pressure, Hydraulic System	PHS	0/4500	PSIA	YES	-	-	Strain Gage Bridge	108.0	202.0	100	20 Hz	1, 2
Differential Pressure, Pitch Actuator	PDA-D	-7000/7000	PSID	YES	-	-	Strain Gage Bridge	197.0	285.0	100	20 Hz	1, 2
Differential Pressure, Yaw Actuator	PYA-D	-7000/7000	PSID	YES	-	-	Strain Gage Bridge	197.0	285.0	100	20 Hz	1, 2
Temperature, Oxidizer Suction	ToS	0/100	°F	YES	-	-	Resistance-Bridge	1.75	5.5	20	4 Hz	1, 2
Temperature, Fuel Suction	TfS	0/100	°F	YES	-	-	Resistance-Bridge	1.75	5.5	20	4 Hz	1, 2
Temperature, Fuel Pressurant Orifice Inlet	TfPOI	0/500	°F	YES	-	-	Resistance-Bridge	8.75	12.47	20	4 Hz	1, 2
Temperature, Oxidizer Pressurant Orifice Inlet	ToPOI	0/500	°F	YES	-	-	Resistance-Bridge	8.75	12.47	20	4 Hz	1, 2
Temperature, Nozzle Extension Skin - 1	TNES-1	0/1000	°F	YES	-	-	T/C & Ref. Comp.	22.2	27.0	20	4 Hz	1, 2
Temperature, Nozzle Extension Skin - 3	TNES-3	0/1000	°F	YES	-	-	T/C & Ref. Comp.	22.2	27.0	20	4 Hz	1, 2
Temperature, Nozzle Extension Skin - 5	TNES-5	0/1000	°F	YES	-	-	T/C & Ref. Comp.	22.2	27.0	20	4 Hz	1, 2
Temperature, Gearbox Bearing No. 6A	TGB-6A	0/500	°F	NO	YES	-	Resistance-Bridge	8.75	12.47	200	40 Hz	1, 4
Temperature, Gearbox Bearing No. 6B	TGB-6B	0/500	°F	NO	YES	-	Resistance-Bridge	8.75	12.47	200	40 Hz	1, 4
Detector, Engine Compartment Fire/Leakage	DECF/DgEC	*	*	NO	-	YES	*	*	*	*	*	* 9
Thrust Chamber Pressure Switch A	$TCPS_A$	ON/OFF	Press.	YES	-	-	Pressure Switch	N/A	N/A	100	0.1 sec	5
Thrust Chamber Pressure Switch B	$TCPS_B$	ON/OFF	Press.	YES	-	-	Pressure Switch	N/A	N/A	100	0.1 sec	5
Thrust Chamber Pressure Switch C	$TCPS_C$	ON/OFF	Press.	YES	-	-	Pressure Switch	N/A	N/A	100	0.1 sec	5
Speed Turbine	NT	0/31000	RPM	YES	-	-	Magnetic Probe & AC to DC Converter	808	929	200	40 Hz	1, 2
Level, Hydraulic Reservoir	LHS	0/100	%	YES	-	-	Potentiometer	I	2.74	20	4 Hz	1, 2

TABLE IV-14 (CONTINUED)
PROPULSION SENSORS

CORE LRE MEASUREMENTS (CON'T)	SYMBOL	RANGE & UNITS		USED ON PREVIOUS TITAN III VEHICLES	TECHNOLOGY EXISTS	NEW TECHNOLOGY REQUIRED	TRANSDUCER TYPE	ACCURACY IN EQUIVALENT ENGINEERING UNITS (±): TRANSDUCER ONLY	ACCURACY IN EQUIVALENT ENGINEERING UNITS (±): TOTAL INSTRUMENTATION SYSTEM	A/B TLM MEAS RESPONSE: SAMPLE RATE IN SAMPLES PER SECOND	A/B TLM MEAS RESPONSE: EQUIVALENT MAXIMUM RESPONSE	APPLICABLE NOTES
Position, Oxidizer Prevalve	LoPV	ON/OFF	Open/ Closed	YES	-	-	Microswitch	N/A	N/A	N/A	N/A	5
Position, Fuel Prevalve	LfPV	ON/OFF	Open/ Closed	YES	-	-	Microswitch	N/A	N/A	N/A	N/A	5
Position, Pitch Actuator	LPA	-5/5	DEG	YES	-	-	Potentiometer	.089	.18	100	20 Hz	1, 2
Position, Yaw Actuator	LYA	-5/5	DEG	YES	-	-	Potentiometer	.089	.18	100	20 Hz	1, 2
CORE TANK MEASUREMENTS	SYMBOL											
Pressure, Oxidizer Tank Gas-1	PgOT-1	0/50	PSIA	NO	YES	-	Potentiometer	1.0	1.5	40	8 Hz	1, 3
Pressure, Fuel Tank Gas-1	PgFT-1	0/50	PSIA	NO	YES	-	Potentiometer	1.0	1.5	40	8 Hz	1, 3
Pressure, Oxidizer Tank Gas-2	PgOT-2	0/50	PSIA	NO	YES	-	Potentiometer	1.0	1.5	40	8 Hz	1, 3
Pressure, Fuel Tank Gas-2	PgFT-2	0/50	PSIA	NO	YES	-	Potentiometer	1.0	1.5	40	8 Hz	1, 3
Temperature, Oxidizer Tank Liquid	ToT	20/200	°F	YES	-	-	Resistance-Bridge	2.5	N/A	N/A	N/A	6, 7
Temperature, Fuel Tank Liquid	TfT	20/200	°F	YES	-	-	Resistance-Bridge	2.5	N/A	N/A	N/A	6, 7
SRM MEASUREMENTS	SYMBOL											
Pressure, Motor Head End, A	$PgMHE_A$	0/1000	PSIA	YES	-	-	Strain Gage Bridge	23.4	31.0	400	80 Hz	1, 8
Pressure, Motor Head End, B	$PgMHE_B$	0/1000	PSIA	YES	-	-	Strain Gage Bridge	23.4	31.0	400	80 Hz	1, 8
Pressure, Injectant Manifold, A	$PoIM_A$	0/1500	PSIA	YES	-	-	Strain Gage Bridge	35.0	43.5	400	80 Hz	1, 8
Position, Injectant Valve, Average Quad 1, A	$LIVA1_A$	0/9.5	VDC	YES	-	-	Potentiometer	.1	.2	100	20 Hz	1, 8
Position, Injectant Valve, Average Quad 1, B	$LIVA1_B$	0/9.5	VDC	YES	-	-	Potentiometer	.1	.2	100	20 Hz	1, 8
Position, Injectant Valve, Average Quad 2, A	$LIVA2_A$	0/9.5	VDC	YES	-	-	Potentiometer	.1	.2	100	20 Hz	1, 8
Position, Injectant Valve, Average Quad 2, B	$LIVA2_B$	0/9.5	VDC	YES	-	-	Potentiometer	.1	.2	100	20 Hz	1, 8
Position, Injectant Valve, Average Quad 3, A	$LIVA3_A$	0/9.5	VDC	YES	-	-	Potentiometer	.1	.2	100	20 Hz	1, 8
Position, Injectant Valve, Average Quad 3, B	$LIVA3_B$	0/9.5	VDC	YES	-	-	Potentiometer	.1	.2	100	20 Hz	1, 8
Position, Injectant Valve, Average Quad 4, A	$LIVA4_A$	0/9.5	VDC	YES	-	-	Potentiometer	.1	.2	100	20 Hz	1, 8
Position, Injectant Valve, Average Quad 4, B	$LIVA4_B$	0/9.5	VDC	YES	-	-	Potentiometer	.1	.2	100	20 Hz	1, 8
Detector, SRM Burnthrough	DBT	*	*	NO	-	YES	*	*	*	*	*	* 9
Position, SRM S/A Device	L-S/A	ON/OFF	Safe/ Armed	NO	YES	-	Rotary Switch Con.	N/A	N/A	N/A	N/A	5, 6, 7
Pressure, Injectant Tank, Gas	PgIT	0/1500	PSIA	YES	-	-	Strain Gage Bridge	35.0	N/A	N/A	N/A	6, 7



ENGINE MODULE TYPICAL OF 5 PLACES

Figure IV-16 Main Propulsion System Measurements

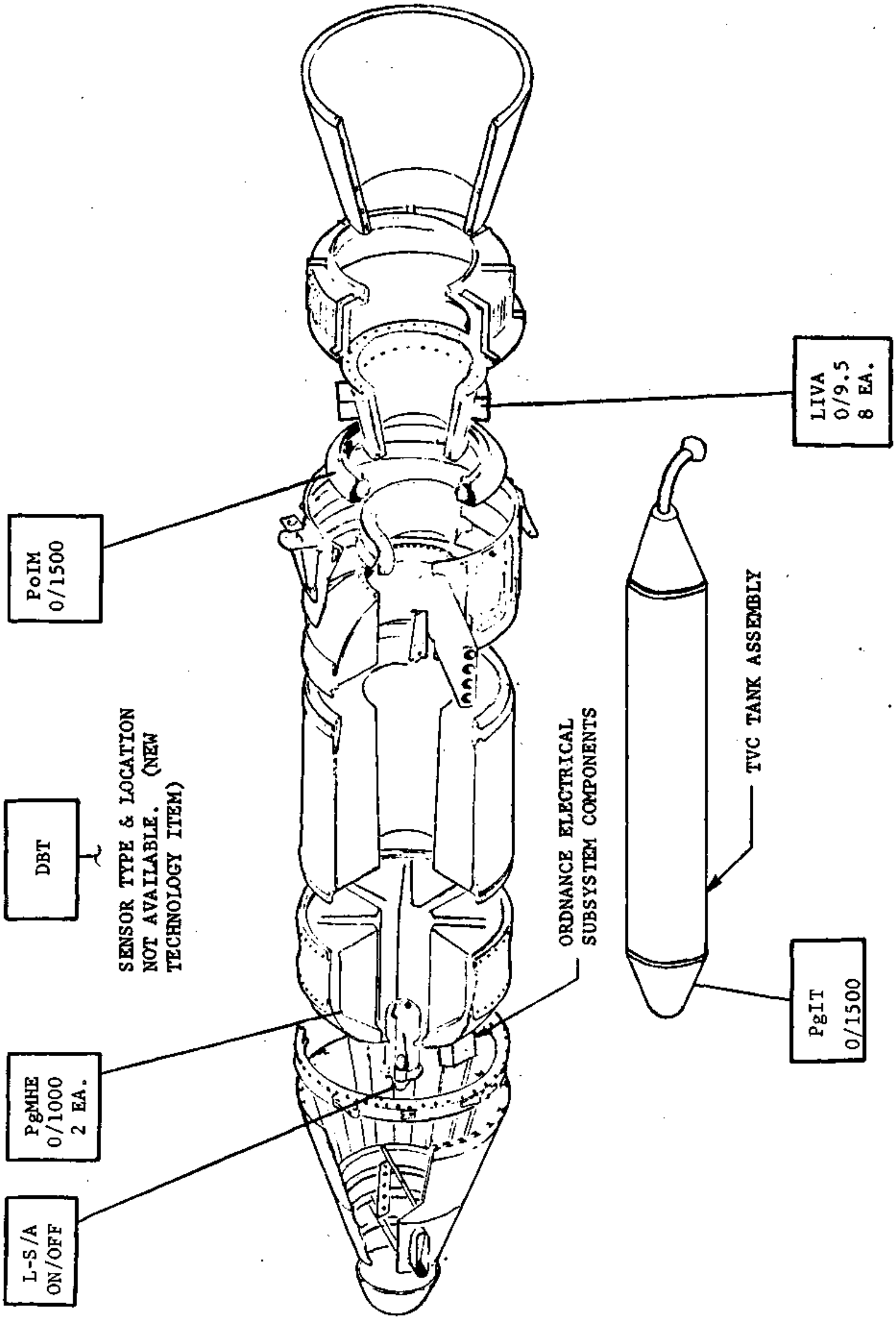


Figure IV-17 Solid Rocket Motor Measurements

2. Ground Support Equipment: The Titan III L operational flow is depicted in Figure IV-18. The checkout and testing activities at the subsystem, system and vehicle level are indicated in the figure. The associated test equipment is identified in Table IV-15. The following paragraphs present an overview of the countdown operations and the electrical launch support and checkout equipment to show the relationship of that equipment to the propulsion systems during launch operations. The launch control system is divided between the Launch Umbilical Tower (LUT) equipment room and the Firing Control Room at the VAB as indicated in Figure IV-19. The launch sequence consists of an R-Count which starts at approximately T-28 hours and a T-Count which starts at T-195 minutes and terminates at T-0. The vehicle is brought to a state of readiness during the R-Count via manual checks and operations under the direction of the pad test conductor. During the R-Count, the propellant tanks are loaded and pressurized under the control of the Propellant Transfer and Pressurization Control Set (PTPCS), and the ordnance is installed. The propellant tank vents are removed near the beginning of the T-Count, T-195 minutes. During the T-Count at approximately T-45 minutes, the final flight controls check is conducted utilizing the vehicle checkout set (VECOS). This equipment verifies the proper operation of the flight controls system and the propulsion system gimbal actuators and thrust vector control valves by applying discrete input commands and monitoring for the proper response. The VECOS automatically controls the test by means of a tape programmer. The T-Count is semi-automatic consisting of a number of manual functions under clock control until T-35 seconds at which time the sequence is completely automatic.

The Control Monitor Group (CMG) controls the time- and event-based countdown for the T-III L booster. The CMG is under direct control of the Launch Control Console (LCC) via the Data Transmission System (DTS). The CMG receives command signals from the LCC for commencing, resetting, holding, and resuming the countdown. The CMG sends signals to the LCC indicating that signals have been received and certain actions have been taken, and to identify holds. The CMG has the capability to issue control functions, monitor launch functions, provide hold, kill and shutdown capability during the launch sequence, reset the system, patch input and output signals, provide simulation signals during combined systems test, and drive the countdown readout indicators at the pad and in the Firing Control Room.

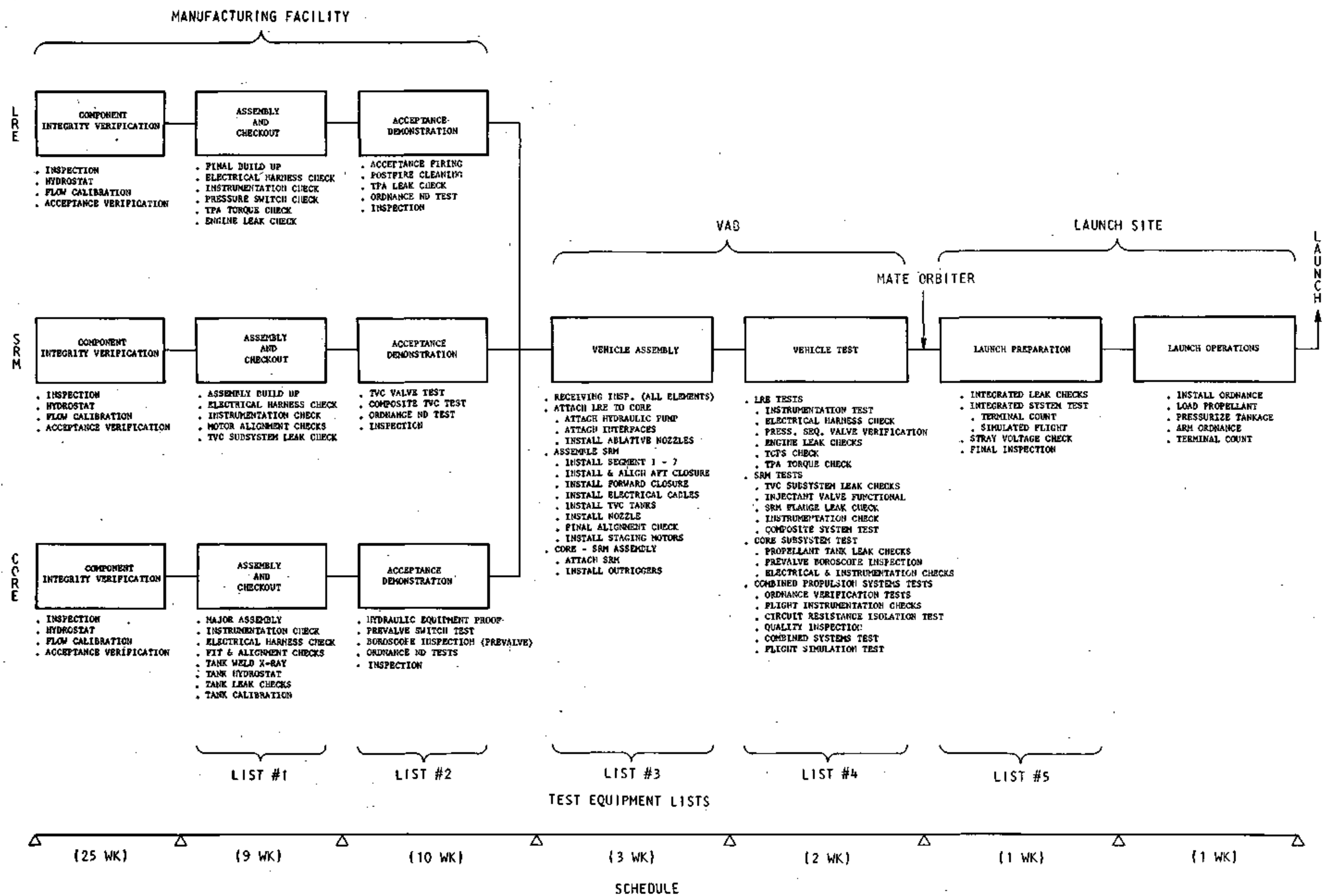


Figure IV-18 TITAN IIIL OPERATIONAL FLOW

TABLE IV-15

PROPULSION TEST EQUIPMENT

TEST EQUIPMENT	USE	EQUIPMENT LIST*
LRE		
Rocket Engine Test Set	The Rocket Engine Test Set is used to accomplish the following: Leak Tests • Propellant Systems • Turbopump Seals • Hot Gas Systems Functional Tests • Thrust Chamber Pressure Switches (TCPS) • Thrust Chamber Valves (TCV) Electrical Continuity Tests • Start Cartridge Squib Wiring • TCV Pressure Sequencing Valve (PSV) Override Solenoids • Thrust Chamber Pressure Switches Electrical Resistance Tests • Start Cartridge Squibs Circuit Wires to Engine Frame Ground • TCPS to Engine Frame Ground • TCVPSV Solenoid to Engine Frame Ground	1, 2, 4
Instrumentation Test Set	The test set is used to test temperature transmitters, pressure transmitters, thermocouples, frequency-to-dc converters, thrust chamber valve potentiometers, and position indicators. The test set performs two insulation resistance checks, continuity/resistance check, zero stimulus check, 50% or 75% stimulus checks and a frequency check.	1, 4
Ordnance Test Set	This portable test set is comprised of ordnance test equipment and an adapter cable, and is used to checkout ordnance devices, ordnance wiring and to check for stray voltages prior to installation of live devices.	2, 4, 5
Turbopump Pressure Leak Test Kit	Verify integrity of turbopump seals.	2
SRM		
Electromechanical Valve Test Set	Verify TVC EMVs are functioning properly, and do not leak.	1

*See Figure IV-18

TABLE IV-15

PROPULSION TEST EQUIPMENT (CONTINUED)

TEST EQUIPMENT	USE	EQUIPMENT LIST*
TVC Test Kit	Verify TVC EMVs are functioning after being installed in the manifold, and/or to the SRM.	2, 4
Critical Circuit Verification Test Set	Verify the integrity of SRM electrical circuits.	1, 4, 5
TVC System Pressure Leak Test Kit	Establish the integrity of all components in the TVC system.	4, 5
SRM Pressure Leak Test Kit	Check for leaks at segment, head end, and aft end closure flanges of the assembled SRM.	4
Instrumentation Test Set	Check operation and calibration of SRM instrumentation.	1, 4
Ordnance Test Set	Nondestructive testing of SRM ordnance and ordnance circuits.	2, 4
CORE		
Electrical Wiring Test Kit	Verify continuity and resistance of all wiring.	1, 4, 5
Prevalve Test Kit, Including Boroscope	Verify prevalve functions open/close, verify position before prop. loading, and check for foreign material on top of prevalve.	2, 4
Hydraulic Control Unit	This item of protable ground equipment is used to perform fill, flush and bleed operations on the core hydraulic system. The HCU provides the means for stroking the core actuators, for commanding the VPDS to operate the core electrically driven hydraulic pumps, and for monitoring hydraulic pressure, hydraulic system reservoir level, and hydraulic accumulator precharge pressure.	4, 5
COMBINED SYSTEMS TEST EQUIPMENT		
Electrical Test Set	Verify the integrity of all SRM to core and core to engine interfaces and the complete booster electrical system.	4, 5

*See Figure IV-18

TABLE IV-15

PROPULSION TEST EQUIPMENT (CONTINUED)

TEST EQUIPMENT	USE	EQUIPMENT LIST*
Ordnance Test Set	This portable test set is composed of ordnance test equipment and an adapter cable, and is used to checkout ordnance devices, ordnance wiring, and to check for stray voltages prior to installation of live devices.	4, 5
Instrumentation Test Set	Used to test temperature and pressure transmitters, thermocouples, frequency converters, thrust chamber valve potentiometers and position indicators. The test set also checks insulation resistance, and continuity/resistance. It can also be used to step calibrate.	4, 5

*See Figure IV-18

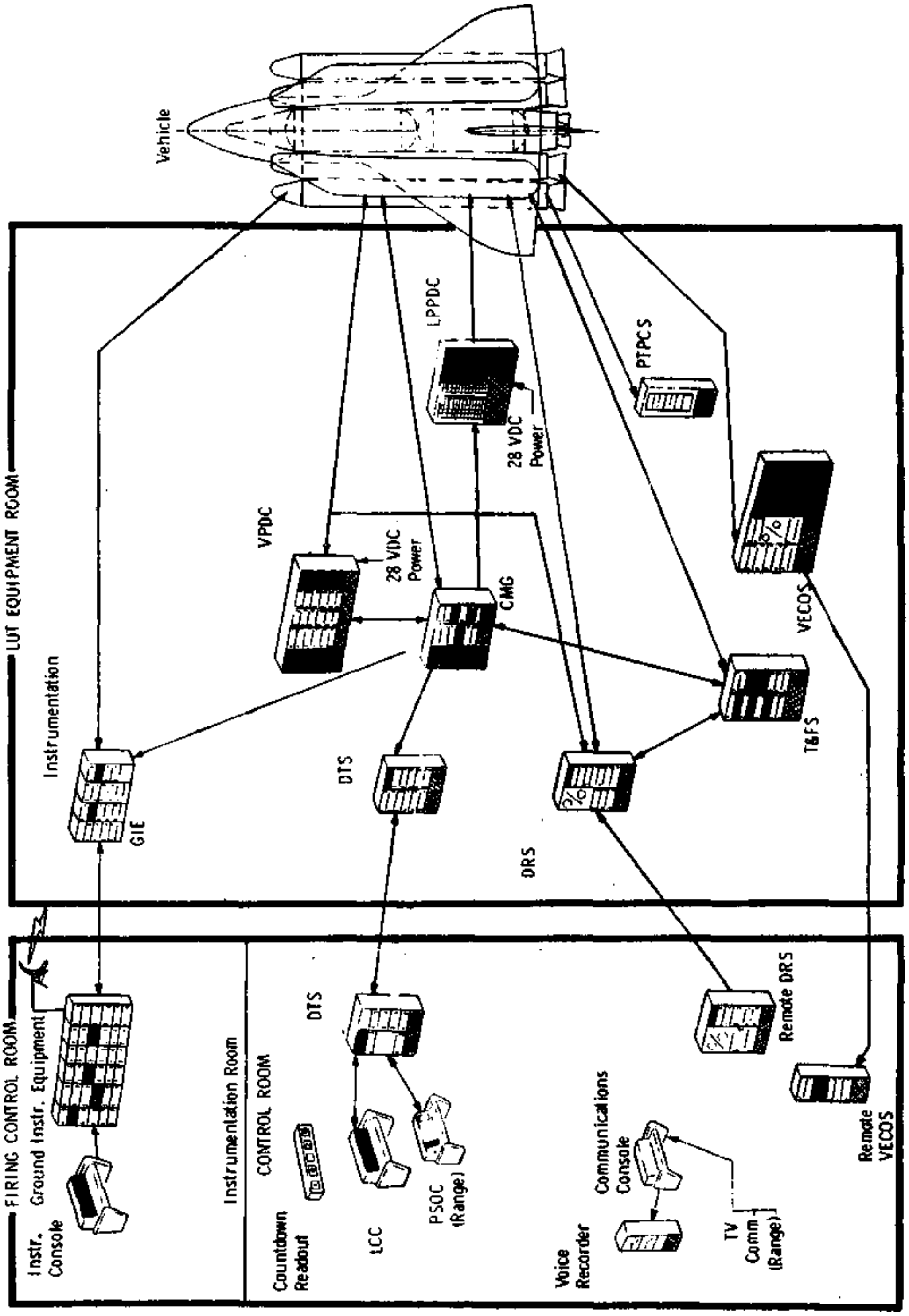


Figure IV-19 Electrical Ground Support Equipment

During the T-Count, power is supplied to the SRM squib firing circuits when the pad safety officer installs red arming plugs in the Van Power Distribution Control (VPDC) rack.

The Data Recording Set (DRS) is initialized during the T-Count. The DRS records the time and change of state of discrete signals and events during the launch sequence. It receives timing information from the CMG. The DRS is used for troubleshooting and fault isolation of countdown sequence malfunctions. A remote DRS is provided in the Firing Control Room.

At approximately T-2 minutes in the count, all instrumentation recorders are started. Data from these recorders is subsequently used for postflight data evaluation of the booster telemetry and landline instrumentation measurements. The recorders are part of the Ground Instrumentation Equipment (GIE) which is located in the Instrumentation Room adjacent to the Firing Control Room in the VIB. In addition to recorders, the instrumentation in the VIB consists of a Control Console, receiving antenna, receivers, decoding equipment and patching capability. The GIE in the instrumentation room interfaces with GIE located in the LUT. The GIE in the LUT consists of signal conditioners, patching and encoding for landline measurements, and voltage controlled oscialIators to modulate signals for transmission to the instrumentation room.

At T-35 seconds, the automatic terminal count is initiated. From this point through ignition of the SRMs, sequencing of events is controlled by the CMG which issues the required commands at preprogrammed times and assesses feedback signals to be considered as prerequisites for subsequent commands.

At approximately T-20 seconds the CMG commands that the SRM ignition circuits be tested. This command causes a test current of low amperage to be sent through the ignition circuits to verify proper installation of the live ordnance device. Stray voltage detectors (SVDs) in the SRM igniters trip as a result of the test and light SVD monitor lights on the flight safety rack. The SVD monitors are then reset via the CMG.

At approximately T-15 seconds the CMG initiates the countdown steering test which is the final prelaunch test of the Guidance and Control Equipment. The CMG verifies that the LRE actuators and TVC valves move off null during this test.

At approximately T-10 seconds the CMG commands and verifies the arming of all ordnance.

At T-3 seconds the CMG initiates the Liquid Rocket Engine (LRE) start sequence. The closure of all thrust chamber pressure switches (TCPS) on the LREs provides the CMG with the indication that LRE chamber pressure is sufficient to permit ignition of the SRMs at approximately T-0.25 seconds.

3. Safety Requirements: Prior to all ordnance installations a Standard Ordnance Circuit Verification Unit (SOCVU) is used during simulation tests to verify ordnance circuits. During installation, an Ordnance Item Test Set (OITS) is used by the installer to verify that the circuits are safe for ordnance connection. A description of the ordnance checkout tests is presented in Table IV-16.

   One of the major tasks of the ordnance system is to start the Solid Rocket Motors. This is accomplished by the SRM igniter assembly which is composed of three major components (Figure IV-20): the safe and arm device, initiator and the main igniter charge. The main igniter charge provides sufficient hot gases to ignite the SRM propellant. The initiator provides the necessary ignition step between the safe and arm device and the larger igniter. The safe and arm device is the control and firing unit for the initiator/igniter assembly. The safe and arm device is mounted in the igniter top boss and contains dual ignition squibs. The squibs are fired by a 28 VDC charge and ignite at 4.5 amps. The 28 volt charge to the squibs originates at the Transient Power Supply (TPS) bus in the core and is switched to the squibs via a Squib Firing Circuit (SFC) in the core vehicle.

   The major components of the safe and arm device used in the SRM ignition train are shown in Figures IV-21 and IV-22. Figure IV-23 is an electrical schematic of the unit and the connecting circuitry. This device incorporates the following safety features: a housing which interferes with easy access of the squibs whether the device is in the safe or arm position (Figure IV-22). This housing also prevents the output from the squibs from reaching the SRM igniter unless the squibs are directed at this window only when the device is in the arm position (Figure IV-21); if the squibs inadvertently fire while in the safe position, the output will be dissipated within a free space of sufficient volume to prevent rupture of the window. To check electrical continuity a squib simulator (resistor-switch) is connected to the firing circuit when the device is in the safe position. To fire the squibs, the squib retainer must be rotated 120 degrees so that the squibs face the blowout window (Figure IV-21). The arming signal rotates the squibs to the firing position. After approximately 105 degrees of rotation the circuit is complete; rotation ceases when the shaft has turned 120 degrees. When the squibs are in the safe position they are disconnected from the firing circuit and are short circuited. An externally visible letter, A for armed and S for safe, are aligned with the safe-arm mechanism to indicate device status (Figure IV-22). Also, internal shaft operated

TABLE I[illegible]6

ORDNANCE CHECKOUT SEQUENCE

PERFORMED TIME	DESCRIPTION OF TEST OR TASK	PURPOSE OF TEST OR TASK
1. Pre-CST	Prior to the Combined Systems Test (CST) the SOCVUs and inert initiators are installed in the SRM.	Preparation for ordnance circuit verification.
2. During CST		
a) T-19	During the CST countdown a CMG signal is sent to test SRM ignition circuits. At this time a low current (<1 amp) is sent via the SFC to the inert initiator in the SRM.	To verify operation of the SVD and the SOCVU under-current sensor
b) T-20 - T-15	SVD reset applied	SVD & SOCVU operation is monitored by DRS
c) T-3	Following the low current (>500, <900 ma) test the inert initiator is cycled from the safe to the arm position.	To prepare for check of full ignition current
d) T-0	The full ignition current (>5 amps) is sent via the core SFCs* to the inert initiator. This current trips the SOCVU.	To verify sufficient current vs time is supplied to "fire" the igniter.
3. Following CST	SOCVU is removed and the inert initiator is replaced with the live S/A. SVDs & SOCVUs are calibrated to verify operation levels. SVDs are reset. DRS data is checked to verify proper operation of SVDs during CST.	Preparations for launch as indicated in task description.
4. During launch C/D		
a) T-30 min	When all other personnel have left pad area, "RED" arm plug is installed on TPS control power distribution rack by pad safety personnel.	To allow test and fire signals to be applied during C/D.
b) T-20 sec	The low test current (<1 amp) is applied to the S/A.	To verify continuity of firing circuit and operation of SVDs prior to arming the igniter.

TABLE IV-16

ORDNANCE CHECKOUT SEQUENCE (CONTINUED)

PERFORMED TIME	DESCRIPTION OF TEST OR TASK	PURPOSE OF TEST OR TASK
c) T-20 to -15	SVD reset applied.	
d) T-10	Igniter is armed	
e) T=0	Igniter is fired if firing lines are not inhibited by pad safety.	
	* Redundant squib firing circuits are used in the core to ignite redundant bridge wires in the S/A device.	

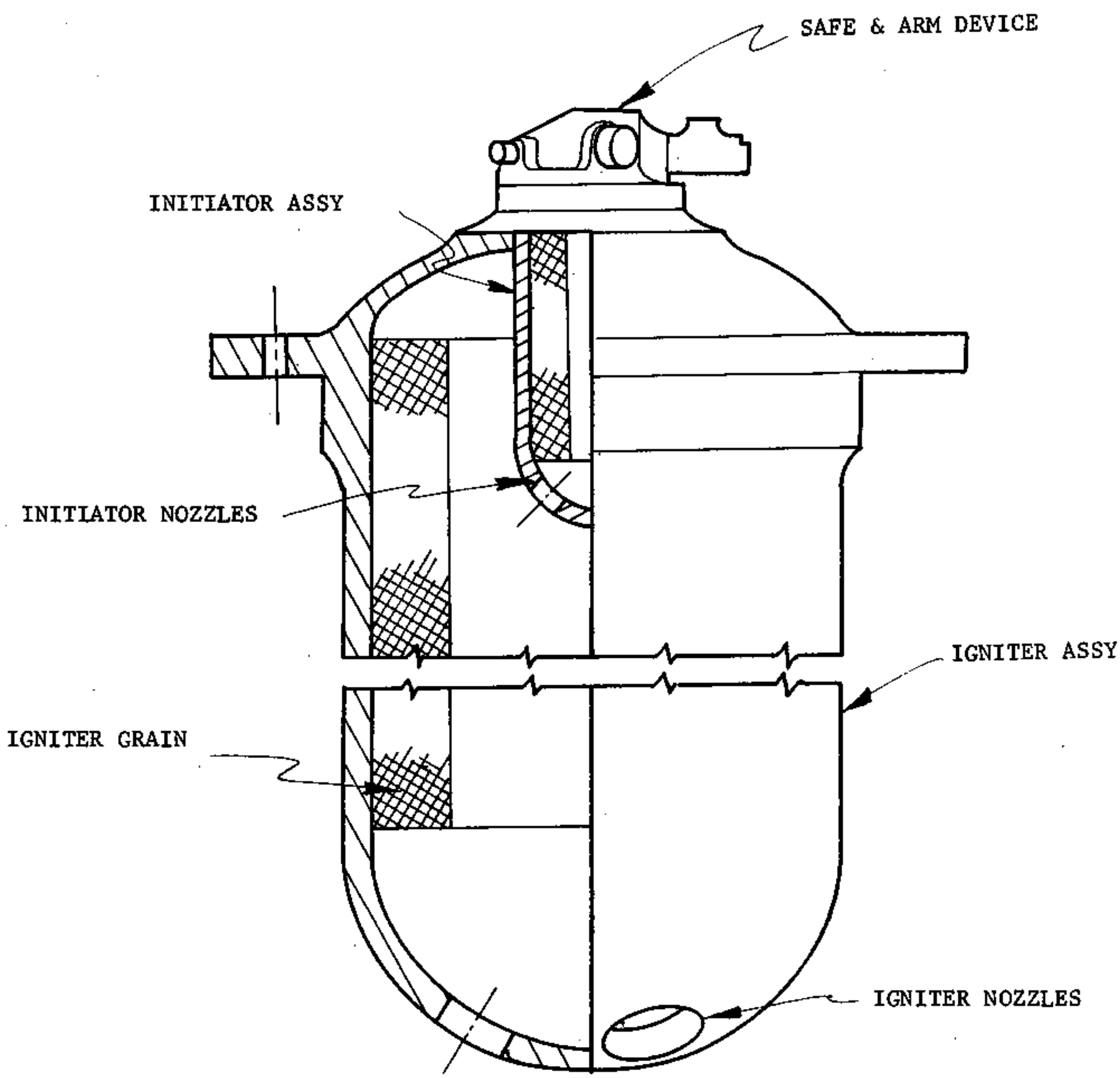


Figure IV-20 SRM Igniter Assembly

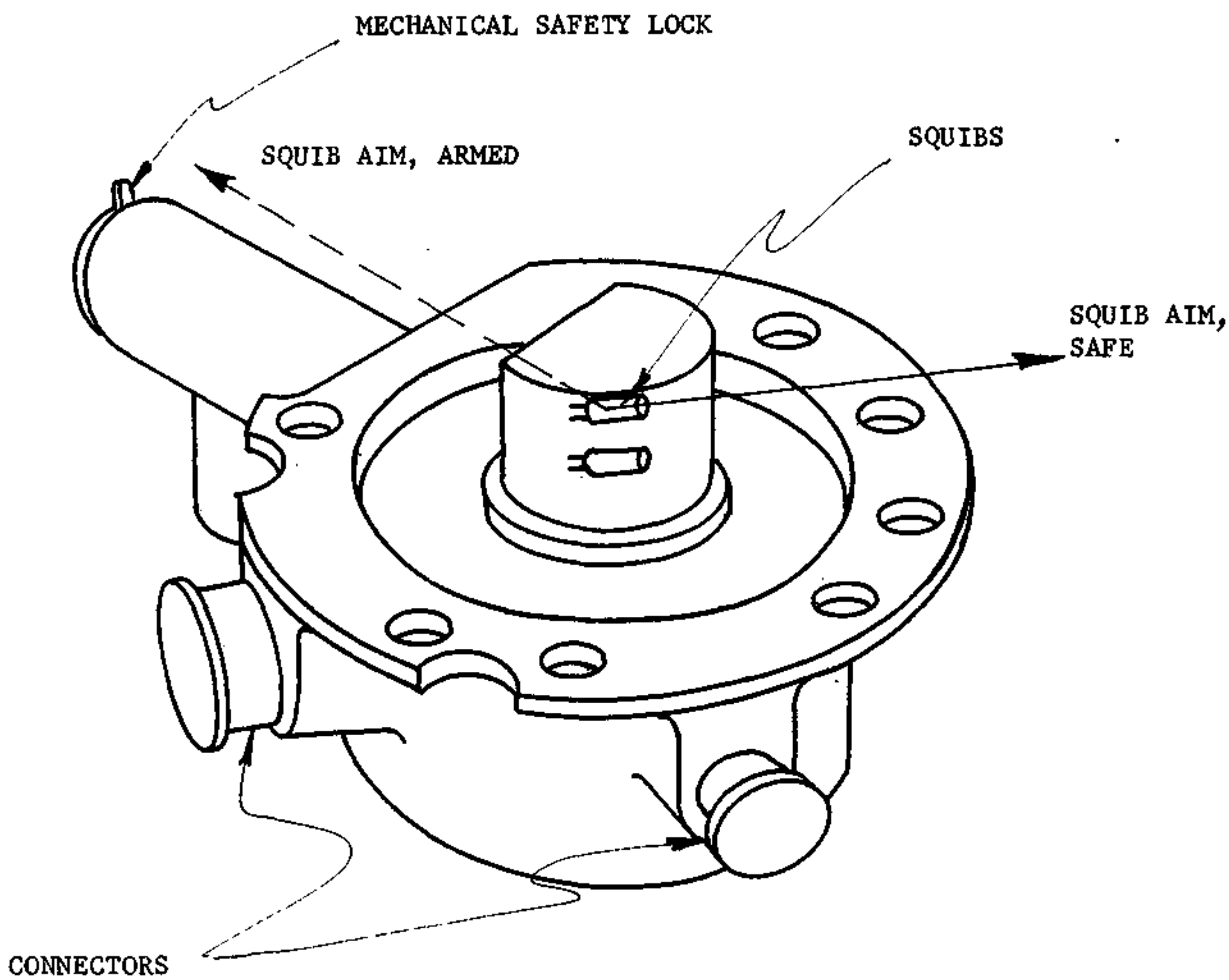


Figure IV-21 Safe and Arm Device

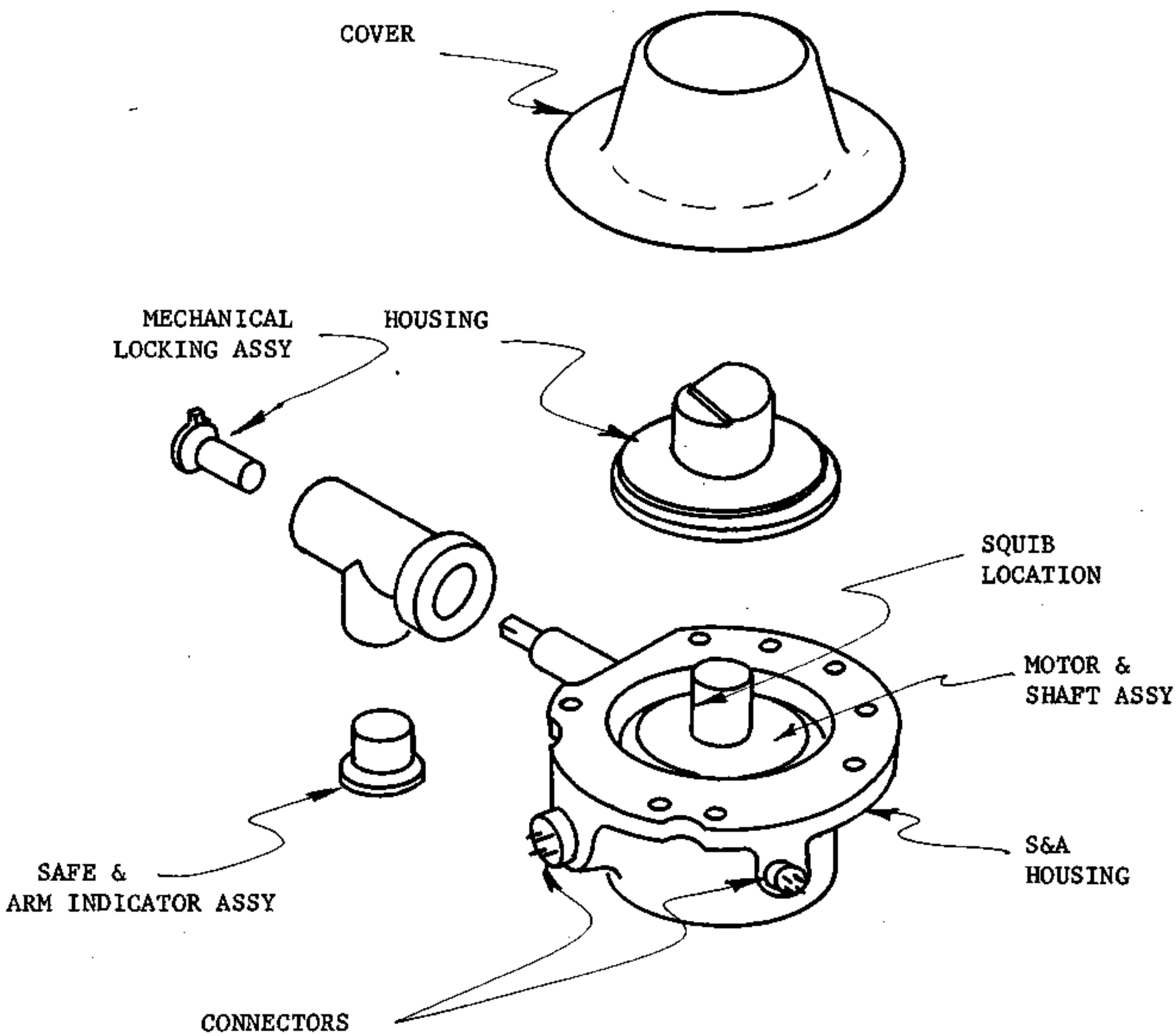
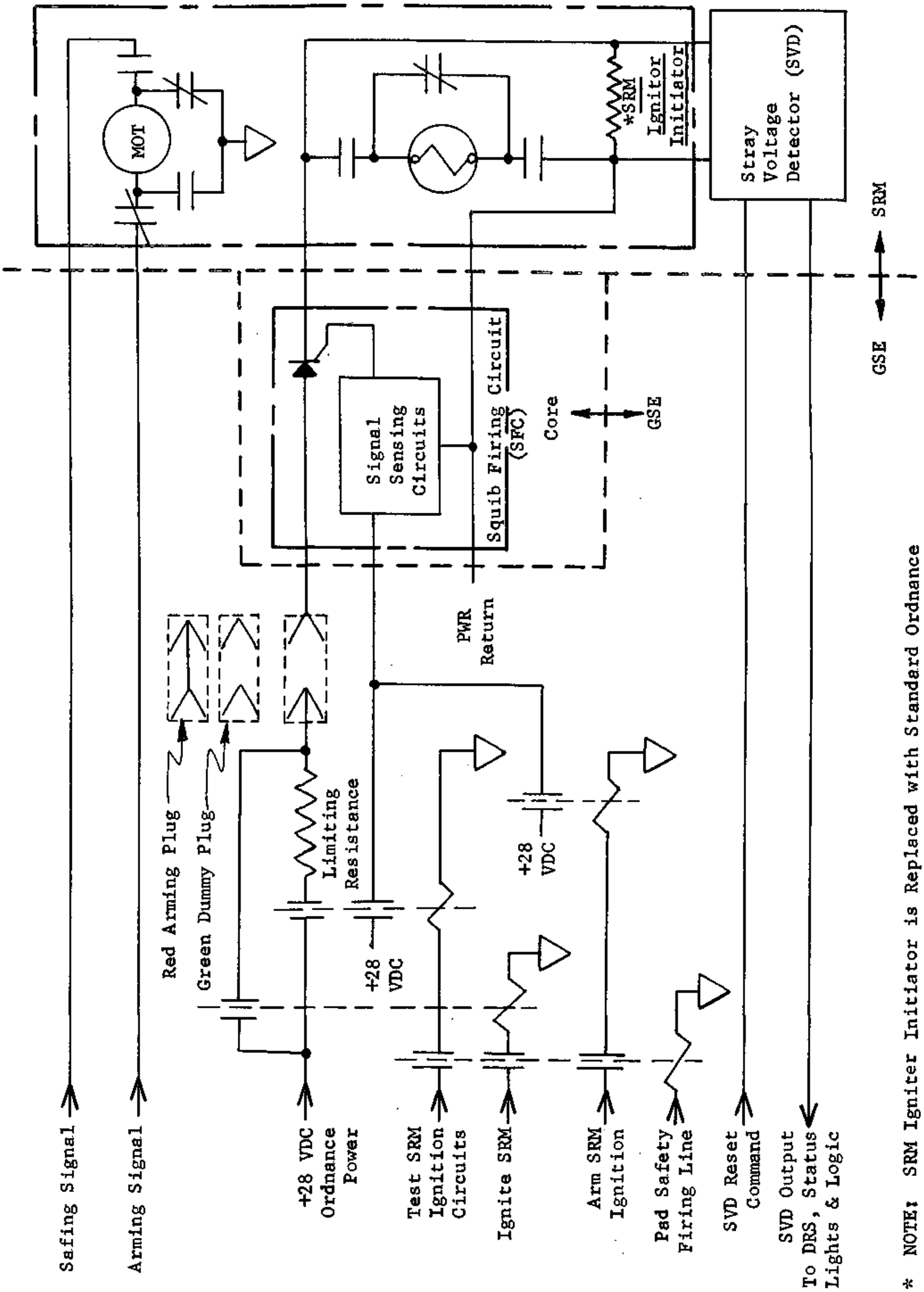


Figure IV-22 Dismantled Safe and Arm Device



* NOTE: SRM Igniter Initiator is Replaced with Standard Ordnance Circuit Verification Unit (SOCVU), and Inert Initiator Until Live Ordnance Installation.

Figure IV-23 SRM Ignition Circuitry

switches permit remote indication of device status. A manually engaged lock prevents actuation of the device (Figure IV-22). Once this lock is removed the unit can only be armed electrically; however, the device can be either electrically or manually actuated, to the safe position. A temperature control switch prevents the overheating of the squib simulator resistors and prevents inadvertant firing. The device and its housing retains the SRM igniter pressures.

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APPENDIX A TITAN III-L FMEA

The objectives of the failure modes and effects analysis (FMEA) were to determine the potential propulsion failure modes and resulting effects on the subsystem, system, and mission; and establish the criticality of the failure and identify candidate failure detection methods for each failure mode to aid in establishing propulsion system measurement requirements. The groundrules and approach used in conducting this analysis are described on page A-2. This appendix contains the FMEA analysis sheets.

FMEA GROUND RULES

1. The FMEA was conducted for the operational mode only.

2. The FMEA was conducted on all components identified in the Titan III L propulsion system definition. (Structural members which perform no function other than providing structural integrity were excluded.)

3. Electrical cables, wiring harnesses and instrumentation were excluded from the FMEA, except where such equipment (such as sensors) were required for control functions within the propulsion subsystem.

4. It was assumed that the proper electrical signal was always transmitted from the control source to the propulsion component requiring such a signal.

5. Human errors were not considered in the failure mode and effects analysis.

6. Failure modes and effects of active and passive thermal protection devices were excluded.

7. Leakage considered in this analysis was categorized to the degree most probable; taking into account the leak path, sealing method, pressure and medium involved in the area under consideration. Leakage requiring a structural failure of the component was not analyzed.

8. Redundant components were identified.

9. The following criticality categories were utilized for potential effect of component failures:

Category	Potential Effect of Failure
1	Loss of life or vehicle
2	Loss of mission
3	All others

a Launch delays were classified as criticality 3.

b The transition point from launch delay, criticality 3 above, to loss of mission, criticality 2, is considered to occur at the ignition of the solid rocket motors.

c The loss of one engine was considered as loss of mission (criticality 2) since the mission requirements and vehicle capability with one engine out were not fully defined.

SYSTEM MAIN PROPULSION PAGE 1

SUBSYSTEM MAIN ENGINE OF 62

COMPONENT INJECTOR

FAILURE MODE AND EFFECTS ANALYSIS

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
1.1.1.1 Injector	Mixes propellant for the combustion process.	Blocked oxidizer orifice (random)	Low	No discernible effect.	None - system capable of tolerating random orifice blockage.	None	None		(3)
		Blocked fuel orifice (random)	Low	Spot erosion of injector face plate.	None - system capable of tolerating random orifice blockage.	None	None		(3)
		Blocked fuel film cooling orifice (random)	Low	Possible tube burnthrough in throat area.	No effect - tube burnthrough will self-cool and prevent spreading of burnthrough area	None	None		(3)

SYSTEM MAIN PROPULSION PAGE 2

SUBSYSTEM MAIN ENGINE OF 62

COMPONENT COMBUSTION CHAMBER

FAILURE MODE AND EFFECTS ANALYSIS

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
1.1.1.2 Combustion Chamber	Contain and direct flow of the combustion gases.	Internal leakage (fuel)	Low	No discernible effect, failure does not propogate.	None	None	None		(3)
		External leakage (fuel)	Low	No discernible effect, failure does not propogate.	Fire hazard	Possible loss of mission resulting from fire aft of engine compartment closure.	Secondary type failures, i.e., loss of instrumentation, particularly the nozzle temperature measurements.		(1)
		External leakage (hot gas - between tube)	Low	No discernible effect, failure does not propogate.	None	None	None		(3)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM MAIN PROPULSION PAGE 3
SUBSYSTEM MAIN ENGINE OF 62
COMPONENT ABLATIVE SKIRT

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
1.1.1.3 Ablative Skirt	Controls expansion of hot gas from chamber to atmosphere producing increased thrust at attitude.	Total or partial loss of skirt before engine shutdown.	Low	Low thrust and/or unbalanced thrust vector.	Low total impulse, possible actuator damage, thrust misalignment and possible damage to adjoining engines.	Possible loss of mission.	Accelerometer measurements, actuator delta P, excessive steering commands.		(1)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM MAIN PROPULSION PAGE 4
SUBSYSTEM MAIN ENGINE OF 62
COMPONENT THRUST CHAMBER PRESSURE SWITCH

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON SUBSYSTEM	FAILURE EFFECT ON SYSTEM	FAILURE EFFECT ON MISSION	RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
1.1.1.4 Thrust Chamber Pressure Switch (3 per engine)	Provides signal to indicate engine operation.	Fail to close on rising thrust chamber pressure (three switches majority vote).	Low	No thrust chamber pressure switch signal from failed switch.	No effect (three switches majority vote).	No effect	Monitor thrust chamber pressure switch signal from each engine.		(3)
		Premature open (three switches majority vote)	Low	Early thrust chamber pressure switch signal from failed switch.	No effect (three switches majority vote).	No effect	Monitor thrust chamber pressure switch signal from each engine.		(3)
		Fail to open on thrust chamber pressure decay (three switches majority vote).	Low	No thrust chamber pressure switch signal.	No effect (three switches majority vote).	No effect	Monitor thrust chamber pressure switch signal from each engine. Majority vote approach recommended.		(3)

SYSTEM MAIN PROPULSION PAGE 5
SUBSYSTEM MAIN ENGINE OF 62
COMPONENT OXIDIZER THRUST CHAMBER VALVE

FAILURE MODE AND EFFECTS ANALYSIS

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
1.1.2.1 Oxidizer Thrust Chamber Valve	Controls oxidizer flow to thrust chamber and gas generator bootstrap line.	External leakage	Low	No effect	Slight loss of oxidizer	No effect	None	Gross leakage would require a structural failure.	(3)
		Internal leakage	Low	No effect	No notable effect	No effect	Visual - prior to start - if protective thrust chamber exit covers are not used.	Gross leakage would require a structural failure.	(3)
		Fails to open during engine start sequence	Low	No thrust chamber combustion.	Engine fails to start.	Launch delay	Chamber pressure, thrust chamber pressure switches, valve trace, possible valve travel limit switch.	Both - the fuel and oxidizer thrust chamber valves would fail to operate due to mechanical linkage.	(3)
		Delayed opening	Low	Possible failure to bootstrap	One engine fails to start	Launch delay	Chamber pressure, thrust chamber pressure switches, valve trace, possible valve travel limit switch.	Both - the fuel and oxidizer thrust chamber valves would fail to operate due to mechanical linkage.	(3)
		Fails to close during engine shutdown sequence.	Low	Would not terminate propellant flow to thrust chamber.	Delayed engine shutdown.	No effect. Flight controls required to compensate for unbalanced force until prevalve closure or complete propellant depletion.	Chamber pressure, thrust chamber pressure switches, valve trace, possible valve travel limit switch.	Both - the fuel and oxidizer thrust chamber valves would fail to operate due to mechanical linkage.	(3)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM MAIN PROPULSION PAGE 6
SUBSYSTEM MAIN ENGINE OF 62
COMPONENT FUEL THRUST CHAMBER VALVE

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
1.1.2.2 Fuel Thrust Chamber Valve	Controls fuel flow to the thrust chamber and gas generator bootstrap line.	External leakage	Low	Fire hazard	Possible loss of system	Possible loss of mission	Aft compartment fire detector		(1)
		Internal leakage	Low	Possible thrust chamber damage during start.	Possible secondary damage to system hardware.	Possible loss of mission.	Aft compartment fire detector.		(1)
		Fails to open during engine start sequence.	Low	No thrust chamber combustion.	Engine fails to start.	Launch delay.	Chamber pressure, thrust chamber pressure switches, valve trace, possible valve travel limit switch.	Both - the fuel and oxidizer thrust chamber valves would fail to operate due to mechanical linkage.	(3)
		Delayed opening	Low	Possible failure to bootstrap.	One engine fails to start.	Launch delay.	Chamber pressure thrust chamber pressure switches, valve trace, possible valve travel limit switch.	Both - the fuel and oxidizer thrust chamber valves would fail to operate due to mechanical linkage.	(3)
		Fails to close during engine shutdown sequence.	Low	Would not terminate propellant flow to thrust chamber.	Delayed engine shutdown.	No effect. Flight controls required to compensate for unbalanced force until prevalve closure or complete propellant depletion.	Chamber pressure, thrust chamber pressure switches, valve trace, possible valve travel limit switch.	Both - the fuel and oxidizer thrust chamber valves would fail to operate due to mechanical linkage.	(3)

SYSTEM MAIN PROPULSION PAGE 7

SUBSYSTEM MAIN ENGINE OF 62

COMPONENT PRESSURE SEQUENCING VALVE

FAILURE MODE AND EFFECTS ANALYSIS

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
1.1.2.3 Pressure Sequencing Valve	Controls fuel pressure to the thrust chamber valve actuator for thrust chamber valve opening and closing.	Fails to shuttle during engine start sequence.	Low	Engine fails to start.	System does not operate.	Launch delay	Chamber pressure, thrust chamber pressure switch, thrust chamber valve limit switch and/or valve trace.		(3)
		Delayed shuttle during engine start sequence.	Low	Engine fails to bootstrap.	System does not operate.	Launch delay.	Chamber pressure, thrust chamber pressure switch, TCV limit switch and/or valve trace.		(3)
		Fails to, or slow to shuttle during engine shutdown sequence.	Low	Delayed thrust chamber valve closing.	Delayed engine shutdown.	No effect. Flight controls required to compensate for unbalanced force until prevalves close or complete propellant depletion.	Thrust chamber valve trace, chamber pressure.	Shutdown assumed to be initiated by propellant exhaustion to one engine.	(3)

SYSTEM MAIN PROPULSION PAGE 8
SUBSYSTEM MAIN ENGINE OF 62
COMPONENT OVERBOARD DRAIN CHECK VALVE

FAILURE MODE AND EFFECTS ANALYSIS

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
1.1.2.4 Overboard Drain Check Valve	Provide atmospheric seal during storage. Bleeds air before engine start. Bleeds fuel for proper thrust chamber valve actuation.	Fail to open during bleed in and start sequence.	Low	Failure to bleed in thrust chamber valve actuator and fuel discharge line	Failure to start or delayed start with thrust overshoot.	Possible launch delay.	None		(3)
		Fails to close at engine start.	Low	No effect.	No effect.	No effect on mission	None	Component leak check required prior to installation.	(3)
		Remains open at engine shutdown.	Low	Slightly higher initial thrust chamber valve closing rate.	No effect.	No effect.			(3)
		Fails to open during shutdown sequence.	Low	Delayed thrust chamber valve closing.	Delayed engine shutdown.	No effect. Flight controls required to compensate for unbalanced force until prevalves close or complete propellant depletion	Thrust chamber valve trace, chamber pressure.	Shutdown assumed to be initiated by propellant exhaustion to one engine.	(3)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM MAIN PROPULSION PAGE 9
SUBSYSTEM MAIN ENGINE OF 62
COMPONENT OXIDIZER PUMP

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
1.1.3.1 Oxidizer Pump	Supply oxidizer to thrust chamber at required pressure and flowrate.	Loss of oxidizer pump discharge pressure and fuel flowrate at start.	Low	Drop in oxidizer pump discharge pressure, drop in thrust chamber pressure.	Fail to start.	Launch delay.	Pump discharge pressure, chamber pressure.		(3)
		Loss of pump discharge pressure during operation.	Low	Drop in oxidizer pump discharge pressure, drop in thrust chamber pressure.	Loss of one engine.	Possible loss of mission.	Pump discharge pressure, chamber pressure.		(2)
		Low oxidizer pump discharge pressure or oxidizer flowrate.	Low	Below normal discharge pressure, low thrust chamber pressure, low engine mixture ratio.	Low performance.	No effect on mission	Pump discharge pressure.		(3)
		External leakage	Low	No effect	No effect due to oxidizer compatibility with aft compartment material.	No effect on mission			(3)

SYSTEM MAIN PROPULSION PAGE 10

FAILURE MODE AND EFFECTS ANALYSIS

SUBSYSTEM MAIN ENGINE OF 62

COMPONENT FUEL PUMP

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON SUBSYSTEM	FAILURE EFFECT ON SYSTEM	FAILURE EFFECT ON MISSION	RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
1.1.3.2 Fuel Pump	Supply fuel to thrust chamber at required pressure and flowrate.	Loss of fuel pump discharge pressure and fuel flowrate at start.	Low	Drop in fuel pump discharge pressure, drop in thrust chamber pressure.	Fail to start.	Launch delay.	Pump discharge pressure, chamber pressure.		(3)
		Loss of pump discharge pressure during operation.	Low	Drop in fuel pump discharge pressure, drop in thrust chamber pressure.	Loss of one engine.	Possible loss of mission.	Pump discharge pressure, chamber pressure.		(2)
		Low fuel pump discharge pressure or fuel flowrate.	Low	Below nominal discharge pressure, low thrust chamber pressure, high engine mixture ratio, possible CC cube damage	Low performance.	No effect on mission	Pump discharge pressure.		(3)
		External leakage.	Low	Possible fire hazard.	Possible fire hazard.	Possible loss of mission.	Aft compartment fire detector.		(1)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM MAIN PROPULSION PAGE 11

SUBSYSTEM MAIN ENGINE OF 62

COMPONENT GEAR BOX

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
1.1.3.3 Gear Box	Transfer turbine power to fuel, oxidizer, lube oil, and hydraulic pumps which supply pressure for the thrust chamber assembly actuators.	Failure to transfer power at start.	Low	Inadequate turbine speed, fuel and oxidizer pump discharge pressure.	Fail to start engine	Delay of mission	Turbine speed, chamber pressure, turbopump discharge pressures.	Recommend TCPS be utilized for engine shutdown.	(3)
		Failure to transfer power during steady state.	Low	Drop in turbine speed, fuel and oxidizer pump discharge pressure. Possible gear box structural failure.	Low performance. Possible system damage resulting from failed gear box.	Possible loss of mission.	Turbine speed, chamber pressure, turbopump discharge pressures.	Recommend TCPS be utilized for engine shutdown.	(1)
		External lube oil leakage (minor).	Low	None	None	None	None	Recommend TCPS be utilized for engine shutdown. Minor external lube oil leakage will not result in gear box damage. Major leakage not considered because the cause requires structural failures.	(3)

SYSTEM MAIN PROPULSION PAGE 12

SUBSYSTEM MAIN ENGINE OF 62

COMPONENT LUBE OIL PUMP

FAILURE MODE AND EFFECTS ANALYSIS

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
1.1.3.4 Lube Oil Pump	Pump oil to cool and lubricate load carrying components in the gear box.	Loss of output or lube oil.	Low	Drop in lube oil flow and an increase in gear box temperature resulting in possible gear box rupture resulting in damage to adjoining engine.	Premature shutdown of one engine. Possible loss of launch vehicle.	Possible loss of mission.	Lube pump discharge pressure, turbine speed, thrust chamber pressure and a thrust chamber pressure switch.		(1)
		External leakage	Low	Fire hazard	Could result in loss of systems.	Possible loss of mission.	Aft compartment fire detector.		(1)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM MAIN PROPULSION PAGE 13
SUBSYSTEM MAIN ENGINE OF 62
COMPONENT LUBE OIL HEAT EXCHANGER

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
1.1.3.5 Lube Oil Heat Exchanger	Extract heat from gearbox lubricating and cooling oil.	Reduced or no fuel flow.	Low	Increase in gearbox temperature resulting in possible gear or bearing damage which could lead to gearbox failure.	Possible premature shutdown of one engine.	Possible loss of mission.	Gearbox and bearing temperature measurement, fuel pressure measurement at cooler outlet. Turbine speed.		(2)
		Reduced or no oil flow.	Low	Increase in gearbox temperature resulting in possible gear or bearing damage which could lead to gearbox failure.	Possible premature shutdown of one engine.	Possible loss of mission.	Gearbox and bearing temperature measurement, fuel pressure measurement at cooler outlet. Turbine speed.		(2)
		External leakage (minor)	Low	None	None	None		Minor lube oil leakage will not result in gearbox damage. Major leakage not considered because the cause requires structural failures.	(3)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM MAIN PROPULSION PAGE 14

SUBSYSTEM MAIN ENGINE OF 62

COMPONENT TURBINE

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON SUBSYSTEM	FAILURE EFFECT ON SYSTEM	FAILURE EFFECT ON MISSION	RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
1.1.3.6 Turbine	Supply power to gear box by converting hot gas energy to mechanical energy.	Inadequate power generation.	Low	Low fuel and oxidizer pump discharge pressure, low turbine speed, resulting in low thrust chamber pressure.	Low performance	Possible loss of mission	Turbine speed, fuel and oxidizer pump discharge pressure or thrust chamber pressure switch.		(2)
		Internal leakage past turbine seal.	Medium	No effect - gear box protected with gear box seal and vent.	No effect	No effect	None		(3)
		External leakage	Low	No effect - subsystem designed to tolerate minor gas leakage.	No effect for anticipated type of seal leakage.	No effect	None		(3)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM MAIN PROPULSION PAGE 15
SUBSYSTEM MAIN ENGINE OF 62
COMPONENT EXHAUST STACK

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
1.1.3.7 Exhaust Stack	Contain the turbine exhaust, and oxidizer fluid heater.	Leakage	Low	No effect, subsystem capable of tolerating flange leakage. Leak area will not propogate.	No effect	No effect	None		(3)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM MAIN PROPULSION PAGE 16

SUBSYSTEM MAIN ENGINE OF 62

COMPONENT GAS GENERATOR

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON SUBSYSTEM	FAILURE EFFECT ON SYSTEM	FAILURE EFFECT ON MISSION	RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
1.1.4.1 Gas Generator	Generate hot gas to drive the turbopumps, supplies fuel tank pressurization gas, heat oxidizer for ox tank pressurization.	Blocked oxidizer orifice, due to contaminants.	Low	Delivers insufficient tank pressurization and low energy for pumps and gear train.	Low performance or premature shutdown of main engine.	Off nominal performance or possible loss of mission.	PcGG, turbine speed, thrust chamber pressure.		(2)
		Blocked fuel orifice due to contaminants.	Low	Delivers insufficient tank pressurization and low energy for pumps and gear train.	Low performance or premature shutdown of main engine, ox rich condition could damage turbine blades.	Off nominal performance or possible loss of mission.	PcGG, turbine speed, thrust chamber pressure.		(2)
		Oxidizer leakage (external)	Low	Delivers insufficient tank pressurization and low energy for pumps and gear train.	Low performance or premature shutdown of main engine.	Off nominal performance or possible loss of mission.	PcGG, turbine speed, thrust chamber pressure.		(2)
		Fuel leakage (external)	Low	Insufficient tank pressurization and low energy to drive pumps and gear train.	Low performance or premature shutdown of main engine. Possible fire hazard.	Possible loss of mission.	PcGG, pump speed, thrust chamber pressure, fuel tank pressure, fire detection system.		(1)
		Hot gas leakage	Low	Possible structural failure of gas generator.	Possible damage to adjoining hardware.	Possible loss of mission due to compounded damage of affected and adjoining engine.	Pressure at turbine inlet, turbine speed, PcGG, and thrust chamber pressure.		(1)

SYSTEM MAIN PROPULSION PAGE 17
SUBSYSTEM MAIN ENGINE OF 62
COMPONENT OXIDIZER CHECK VALVE

FAILURE MODE AND EFFECTS ANALYSIS

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
1.1.4.2 Oxidizer Check Valve	Sequences flow of oxidizer to gas generator. Prevents start cartridge exhaust products from entering boot-strap line.	Fails to open	Low	No oxidizer flow to gas generator.	No gas generator operation.	Delay of mission.	Thrust chamber pressure fails to rise, thrust chamber pressure switch did not close.		(2)
		External leakage	Low	Lower than nominal or no oxidizer flow to gas generator.	Low performance or premature shutdown.	Off nominal performance or loss of mission.	Thrust chamber pressure switch might make.		(2)
		Reverse flow during engine start due to check valve failure to seat.	Low	Start cartridge particles plugging injector orifices. Low gas generator performance.	Off nominal performance. Possible failure of engine to bootstrap.	Launch delay	Thrust chamber pressure, possible thrust chamber pressure switch make.		(2)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM MAIN PROPULSION PAGE 18
SUBSYSTEM MAIN ENGINE OF 62
COMPONENT FUEL CHECK VALVE

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON SUBSYSTEM	FAILURE EFFECT ON SYSTEM	FAILURE EFFECT ON MISSION	RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
1.1.4.3 Fuel Check Valve	Sequences flow of fuel to gas generator. Prevents start cartridge exhaust products from entering bootstrap line.	Fails to open.	Low	No fuel flow to gas generator.	No gas generator operation.	Delay of mission.	Thrust chamber pressure fails to rise. Thrust chamber pressure switch did not close.		(2)
		External leakage	Low	Lower than nominal or no fuel flow to gas generator. Possible fire hazard.	Low performance or premature shutdown.	Off nominal performance or loss of mission.	Thrust chamber pressure switch might make.	Leak would have to be such that the fuel impinges on a hot surface in order to cause a fire hazard.	(1)
		Reverse flow during engine start due to check valve failure to seat.	Low	Start cartridge particles plugging injector orifices. Low gas generator performance.	Off nominal performance. Possible failure of engine to bootstrap.	Launch delay	Thrust chamber pressure build up, no ignition.		(2)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM MAIN PROPULSION PAGE 19
SUBSYSTEM MAIN ENGINE OF 62
COMPONENT OXIDIZER CAVITATING VENTURI

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
1.1.4.4 Oxidizer Cavitating Venturi	Control the oxidizer flow rate into the gas generator.	External leakage	Low	Degradation of gas generator performance.	Possible low performance or premature shutdown of main engine.	Off nominal performance or possible loss of mission.	PcGG, turbine speed, thrust chamber pressure.		(2)
		Plugged venturi	Low	No oxidizer to gas generator.	Loss of one engine.	Possible loss of mission.	PcGG, turbine speed, thrust chamber pressure.		(2)

SYSTEM MAIN PROPULSION PAGE 20

SUBSYSTEM MAIN ENGINE OF 62

COMPONENT FUEL CAVITATING VENTURI

FAILURE MODE AND EFFECTS ANALYSIS

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
1.1.4.5 Fuel Cavitating Venturi	Control the fuel flowrate into the gas generator.	External leakage	Low	Degradation of gas generator performance.	Possible low performance or premature shutdown of main engine. Possible fire.	Possible loss of mission.	PcGG, turbine speed, thrust chamber pressure, fire detection system.		(1)
		Plugged venturi	Low	No fuel to gas generator.	Loss of one engine.	Possible loss of mission.	PcGG, turbine speed, thrust chamber pressure.		(2)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM MAIN PROPULSION PAGE 21
SUBSYSTEM MAIN PRESSURIZATION OF 42
COMPONENT OXIDIZER LINE FILTER

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON SUBSYSTEM	FAILURE EFFECT ON SYSTEM	FAILURE EFFECT ON MISSION	RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
1.1.4.6 Oxidizer Line Filter	Provides protection for the GGA from possible contaminants.	Restricted flow due to contamination.	Low	Lower than nominal or no oxidizer flow to gas generator.	Low performance or premature shutdown due to insufficient turbine speed and tank pressure.	Off nominal performance or loss of mission.	Reduction in PcGG, drop in thrust chamber pressure. Possible thrust chamber pressure switch make.		(2)
		Damaged filter screen allowing passage of contaminants.	Low	Possible plugging of the gas generator injector ports	Low performance or premature shutdown due to insufficient turbine speed and tank pressure.	Off nominal performance or loss of mission.	Reduction in PcGG, drop in thrust chamber pressure. Possible thrust chamber pressure switch make.		(2)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM MAIN PROPULSION PAGE 22

SUBSYSTEM MAIN PRESSURIZATION OF 62

COMPONENT FUEL LINE FILTER

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
1.1.4.7 Fuel Line Filter	Provides protection for the GGA from possible contaminants.	Restricted flow due to contamination.	Low	Lower than nominal or no fuel flow to gas generator.	Low performance or premature shutdown due to insufficient turbine speed and tank pressure. Low fuel flowrate could cause overheating of the gas generator and a burn-through.	Off nominal performance or loss of mission.	Reduction in PcGG, drop in thrust chamber pressure, possible thrust chamber pressure switch make.		(2)
		Damaged filter screen allowing passage of contaminants.	Low	Possible plugging of the gas generator injector ports	Low performance or premature shutdown due to insufficient turbine speed and tank pressure.	Off nominal performance or loss of mission.	Reduction in PcGG, drop in thrust chamber pressure. Possible thrust chamber pressure switch make.		(2)

SYSTEM MAIN PROPULSION PAGE 23

SUBSYSTEM MAIN ENGINE OF 42

COMPONENT START CARTRIDGE

FAILURE MODE AND EFFECTS ANALYSIS

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
1.1.5.1 Start Cartridge	Provide hot gas energy to turbine for engine start.	Low energy	Low	Fuel pump discharge pressure not sufficient to open thrust chamber valves.	Engine fails to start	Delay of mission	Thrust chamber pressure, thrust chamber pressure switch.		(3)
		Insufficient duration	Low	Failure to bootstrap	Premature engine shutdown	Delay of mission	Thrust chamber pressure, thrust chamber pressure switch.		(3)
		High energy or long duration	Low	Thrust chamber pressure spike	Thrust overshoot	No effect on mission	Thrust chamber pressure, pump discharge pressure.		(3)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM MAIN PROPULSION PAGE 24
SUBSYSTEM MAIN ENGINE OF 62
COMPONENT INITIATOR

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
1.1.5.2 Initiator	Convert electrical energy into heat and pressure sufficient enough to ignite the solid start cartridge.	Fail to ignite (dual bridge wire redundant)	Low	Fail to ignite solid start cartridge	Fail to start one subassembly.	Delay of mission	Thrust chamber pressure, transducer and TC pressure switches.		(3)
		Premature ignition	Low	Possible destruction of gearbox, pumps, etc., because of no load and overspeed.	Possible loss of system.	Loss of mission	Turbine overspeed, fire detectors.	The squib firing circuit which supplied the firing current to the start cartridge initiator includes the following interlocks to prevent premature ignition: a) must be armed with 28 VDC and supplied with 28 VDC operate power. b) Must be supplied with Stage I engine start power. c) Must be supplied with an input "Fire" signal of sufficient amplitude and duration to provide switch operation.	(1)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM MAIN PROPULSION PAGE 25

SUBSYSTEM MAIN ENGINE OF 62

COMPONENT GIMBAL BLOCK

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
1.1.6.1 Gimbal Block	Transfer thrust, provides pivot for vehicle pitch roll and yaw.	Failure to pivot due to excessive friction.	Low	Loss of TVC on one engine.	Failure to respond to steering commands	Loss of mission	Actuator position, hydraulic pressure, and actuator delta pressure.	Provide overtravel switches. Presently baselined failed engine must be shutdown.	(2)
		Slow pivot due to excessive friction.	Low	Slow pitch/yaw control.	Slow response to steering commands.	No effect on mission	Actuator position, hydraulic pressure, and actuator delta pressure.		(3)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM MAIN PROPULSION PAGE 26
SUBSYSTEM MAIN ENGINE OF 52
COMPONENT GIMBAL ACTUATOR (2)

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
1.1.6.2 Gimbal Actuator (2)	Positions the thrust chamber for thrust vector control.	Actuator goes hard over and remains there.	Low	Possible structural damage.	Loss of TVC on one engine.	Possible loss of mission.	Actuator position as compared to command (rate), actuator delta pressure.	Probable engine shut-down required when the malfunction is identified.	(1)
		No response of actuator to commands.	Low	Inability to position thrust chamber.	Loss of TVC on one engine.	Possible loss of mission.	Actuator position as compared to command (rate), actuator delta pressure.		(2)
		Unstable actuator positions (oscillations)	Low	None	Oscillation of engine thrust vector.	Oscillatory excitation of vehicle (until actuator can be locked into null then no effect).	Actuator position as compared to command (rate), actuator delta pressure, with measurements analyzed for oscillations.	Lock actuator in null position when this failure mode is detected.	(3)
		Unmodulated actuator position (full extend or full retract positions only)	Low	None	Unstable thrust vector in one plane.	Possible vehicle flight instability (until actuator can be locked in null), then no effect.	Actuator position as compared to command (rate), actuator delta pressure.	Lock actuator in null position when this failure mode is detected.	(3)
		Slow response	Low	Slow movement of engine thrust chamber.	Poor TVC	No effect	Actuator position as compared to command (rate), actuator delta pressure.		(3)
		Null shift	Low	None	Poor TVC	No effect - compensated by flight controls.	Actuator position as compared to command (rate), actuator delta pressure.		(3)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM MAIN PROPULSION PAGE 27
SUBSYSTEM MAIN ENGINE OF 62
COMPONENT HYDRAULIC PUMP

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
1.1.6.3 Hydraulic Pump	Supply hydraulic fluid to actuators at required pressure and flowrate for thrust vector control.	Restricted flow	Low	Slow response to control commands	Poor TVC	No effect on the mission - compensated by flight controls.	Hydraulic pressure, actuator delta pressure, actuator position as compared to command.		(3)
		Loss of hydraulic pressure	Low	Failure to position engine	Loss of TVC for one engine. Possible structural damage to engine.	Possible loss of mission.	Hydraulic pressure, actuator delta pressure, actuator position as compared to command.	Probable engine shutdown required upon detection.	(1)
		External Leakage	Low	Failure to position engine	Loss of TVC for one engine	Possible loss of mission.	Hydraulic pressure, actuator delta pressure, actuator position as compared to command.	Probable engine shutdown required upon detection.	(2)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM MAIN PROPULSION PAGE 28
SUBSYSTEM MAIN ENGINE OF 62
COMPONENT AUXILIARY PUMP

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
1.1.6.4 Auxiliary Pump	To provide hydraulic pressure during the start transient to maintain engine null position.	Low pressure or flow	Low	Possible engine travel during start	Possible engine damage during start if movement is too fast.	Possible launch delay.	Hydraulic pressure indication.	Recommend launch hold predicated on aux. pump failure.	(3)
		Fail to generate pressure and flow-rate.	Low	Possible engine travel during start	Possible engine damage during start if movement is too fast.	Launch delay.	Hydraulic pressure indicator.	Recommend launch hold predicated on aux. pump failure.	(2)
		External leakage	Low	None (if leakage is large enough low pressure could result - effects same as Low Pressure above).	No effect	No effect	Low hydraulic pressure at start.	Recommend launch hold predicated on aux. pump failure.	(3)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM MAIN PROPULSION PAGE 29

SUBSYSTEM MAIN PROPELLANT MGMT. OF 62

COMPONENT OXIDIZER TANK

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
1.2.1.1 Oxidizer Tank	Provide storage of propellants and direct propellants into the engine.	Minor leakage, around flanges or seals.	Low	None, system tolerances allow for minor loss of oxidizer.	None	None		Major leakages in excess of functional system tolerances would require that a structural failure occur.	(3)

SYSTEM MAIN PROPULSION PAGE 30

SUBSYSTEM MAIN PROPELLANT MGMT. OF 62

COMPONENT OXIDIZER POGO SUPPRESSOR

FAILURE MODE AND EFFECTS ANALYSIS

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
1.2.1.2 Oxidizer POGO Suppressor	Act as an accumulator in each oxidizer feedline to decouple engine and airframe longitudinal oscillations.	Gas leakage	Low	Loss of 10% of POGO suppression capability.	No effect	No effect	Oxidizer bellows displacement, oxidizer bellows pressure.		(3)
		External liquid leakage	Low	Minor leakage - no effect	No effect	No effect			(3)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM MAIN PROPULSION PAGE 31
SUBSYSTEM MAIN PROPELLANT MGMT. OF 62
COMPONENT OXIDIZER PREVALVE

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
1.2.1.3 Oxidizer Prevalve	Prevent oxidizer from entering the thrust chamber prematurely and to isolate the oxidizer from the thrust chamber after engine shutdown.	Fails to open	Low	No oxidizer to affected engine.	No effect	Mission hold	Prevalve position indicator.		(3)
		Fails open during ascent.	Low	Loss of one half of engine shutdown capability on one engine module.	No effect	No effect	None		(3)
		External leakage, minor	Low	No effect, subsystem can tolerate this type of leakage.	No effect.	No effect.	None		(3)

SYSTEM MAIN PROPULSION PAGE 32

SUBSYSTEM MAIN PROPELLANT MGMT. OF 62

COMPONENT FUEL TANK

FAILURE MODE AND EFFECTS ANALYSIS

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
1.2.2.1 Fuel Tank	Provide storage of propellant and to direct propellant into the engine.	Minor leakage, around flanges or seals.	Low	None, system tolerances allow for minor loss of fuel.	Possible fire hazard if fuel contacts hot surfaces.	Possible loss of mission.		Major leakages in excess of functional system tolerances would require that a structural failure occur.	(2)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM MAIN PROPULSION PAGE 33
SUBSYSTEM MAIN PROPELLANT MGMT. OF 62
COMPONENT FUEL POGO SUPPRESSOR

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
1.2.2.2 Fuel POGO Suppressor	Act as an accumulator in each fuel feedline to decouple engine and airframe longitudinal oscillations.	Gas bag leakage	Low	Reduced POGO suppression capability	No effect	No effect	None	System can tolerate degraded suppression operation on one engine	(3)
		External leakage	Low	Minor fuel loss	Fire hazard	Possible loss of mission	Flame detector		(1)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM MAIN PROPULSION PAGE 34
SUBSYSTEM MAIN PROPELLANT MGMT. OF 62
COMPONENT FUEL PREVALVE

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
1.2.2.3 Fuel Prevalve	Prevent fuel from entering the thrust chamber prematurely and to isolate the fuel from the thrust chamber after engine shutdown.	Fails to open	Low	No fuel to affected engine.	No effect	Mission hold	Prevalve position indicator.		(3)
		Fails open during ascent.	Low	Loss of one half of engine shutdown capability on one engine.	No effect	No effect	None		(3)
		External leakage, minor	Low	No effect, subsystem can tolerate this type of leakage.	Possible fire hazard if fuel contacts hot surfaces.	Possible loss of mission.		Major leakage in excess of functional system tolerances would require that a structural failure occur.	(2)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM MAIN PROPULSION PAGE 35

SUBSYSTEM MAIN PRESSURIZATION OF 62

COMPONENT FLUID HEATER

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
1.3.1.1 Fluid Heater	Heat liquid N_2O_4 to a vapor state to be used for oxidizer tank pressurization.	Leakage (external)	Low	Low oxidizer autogenous pressure.	Possible 20% reduction in oxidizer tank gas pressure.	None, system capable of operating with reduced tank gas pressure.	Tank gas pressure presently baselined	Would require a structural failure.	(3)
		Restricted or no flow	Low	Low oxidizer autogenous pressure.	Possible 20% reduction in oxidizer tank gas pressure.	None, system capable of operating with reduced tank gas pressure.	Tank gas pressure, pressure at oxidizer orifice inlet, temperature at oxidizer orifice inlet.		(3)

SYSTEM MAIN PROPULSION PAGE 36
SUBSYSTEM MAIN PRESSURIZATION OF 62
COMPONENT CAVITATING VENTURI

FAILURE MODE AND EFFECTS ANALYSIS

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON SUBSYSTEM	FAILURE EFFECT ON SYSTEM	FAILURE EFFECT ON MISSION	RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
1.3.1.2 Cavitating Venturi	Control oxidizer tank gas pressurant flow.	Improper size installed.	Low	Slight degradation of pressurant flowrate.	Possible off nominal oxidizer tank pressurization rate.	None	Oxidizer tank pressure.		(3)
		Not installed	Zero					Lines cannot be connected without the venturi.	
		External leakage	Low	Slight loss of oxidizer.	Failure mode can be tolerated by system.	None			(3)
		Plugged venturi	Low	Loss of oxidizer pressurant from one engine.	Possible 20% reduction in tank pressure, total system can tolerate this condition.	None			(3)

FAILURE MODE AND EFFECTS ANALYSIS

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
1.3.1.3 Backpressure Orifice	Control the residence time of oxidizer in the Fluid Heater.	Incorrect installation, undersized.	Low	Degraded control of oxidizer tank pressurant enthalpy.	Increased autogenous system enthalpy from one engine.	No effect, or off nominal performance (A full 20% increase in tank pressure will not cause a tank failure.)	Pressure and temperature measurement at backpressure orifice inlet. Presently baselined.	Orifices manufactured in sizes which will cause minor changes in gas flow (vernier adjustment).	(3)
		Incorrect installation, oversized.	Low	Degraded control of oxidizer tank pressurant enthalpy.	Decreased autogenous system enthalpy from one engine.	No effect, or off nominal performance (A full 20% decrease in tank pressure will not cause a total loss in tank gas pressure.)	Pressure and temperature measurement at backpressure orifice inlet. Presently baselined.		(3)
		Blocked orifice.	Low	Inability to pressurize fuel tank from one subassembly.	20% reduction in tank gas pressure. (Tolerable for overall system operation.)	Possible reduction in performance.	Oxidizer tank gas pressure.		(3)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM MAIN PROPULSION PAGE 38

SUBSYSTEM MAIN PRESSURIZATION OF 62

COMPONENT BURST DIAPHRAGM (ALRC)

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
1.3.1.4 Burst Diaphragm (ALRC)	Seals oxidizer pressurization assembly prior to operation and facilitates system leak checks.	Premature rupture	Low	No effect	No effect	No effect	None	Diaphragm is coined on both sides to insure against improper installation. Parts are not interchangeable with standard burst disc assemblies.	(3)
		Fail to rupture	Low	No pressurant from one engine assembly	20% reduction in tank pressure.	None or slight reduction in performance.	Tank gas pressure.		(3)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM MAIN PROPULSION

SUBSYSTEM MAIN PRESSURIZATION
COMPONENT CHECK VALVE

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON SUBSYSTEM	FAILURE EFFECT ON SYSTEM	FAILURE EFFECT ON MISSION	RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
1.3.1.5 Check Valve (Oxidizer)	Isolates direction of gas flow from heaters to the tank. Required only in event of engine out condition.	Fails to open	Low	No pressurant flow from one engine.	20% reduction in tank pressure.	None to slight reduction in performance.	Tank gas pressure low, pressure at back pressure orifice.		(3)
		Fails to close, reverse leakage	Low	No effect	No effect	No effect	None		(3)

SYSTEM MAIN PROPULSION PAGE 40

SUBSYSTEM MAIN PRESSURIZATION OF 62

COMPONENT OX PRESSURIZATION & VENT CONNECTOR

FAILURE MODE AND EFFECTS ANALYSIS

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON SUBSYSTEM	FAILURE EFFECT ON SYSTEM	FAILURE EFFECT ON MISSION	RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
1.3.1.6 Ox Pressurization and Vent Connector	Provide means to pre-pressurize or vent oxidizer tanks.	Leakage after tank pressurization and removal of ground half of connector.	Low	Loss of oxidizer tank pressure	Possible launch abort.	Delay	Verify no gas leakage before securing cap.		(3)
		Cap leakage (redundant capped seal)	Low	No effect (requires two failures - connector and cap - to effect system)	No effect	No effect	None	Leakage past redundant system is considered within tolerable limits.	(3)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM MAIN PROPULSION PAGE 41
SUBSYSTEM MAIN PRESSURIZATION OF 62
COMPONENT OXIDIZER TANK BURST DIAPHRAGM MMC

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
1.3.1.7 Oxidizer Tank Burst Diaphragm (MMC)	Seals oxidizer pressurization assembly prior to operation and facilitates system leak checks.	Premature rupture (check valve provides redundant feature).	Low	No effect	No effect	No effect	None		(3)
		Fail to rupture	Zero	No oxidizer pressurant.	Possible insufficient tank gas pressure for engine NPSH, and tank structural integrity.	Possible loss of mission due to tank rupture.	Tank gas pressure, pump inlet pressure.	MMC disc could possibly be eliminated with the use of line check valves. Disc is special assembly hardware which can be installed in only one way. No substitute can be made.	–

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM MAIN PROPULSION PAGE 42

SUBSYSTEM MAIN PRESSURIZATION OF 62

COMPONENT GAS COOLER

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON SUBSYSTEM	FAILURE EFFECT ON SYSTEM	FAILURE EFFECT ON MISSION	RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
1.3.2.1 Gas Cooler	Reduce the temperature of a portion of the gas generator exhaust products before using them for booster fuel tank pressurization.	No fuel flow.	Low	No control of fuel pressurant temperature for the affected subassembly. Possible rupture of gas cooler because of exothermic reaction of trapped fuel.	Possible loss of surrounding components and tank gas pressure. Possible fire hazard.	Possible loss of mission.	Tank gas pressure and temperature, temperature of fuel entering and leaving gas cooler, pressure and temperature at ALRC side of sonic flow control nozzle.		(1)
		Reduced fuel flow	Low	Partial loss of control.	None	None	Tank gas pressure and temperature, temperature of fuel entering and leaving gas cooler, pressure and temperature at ALRC side of sonic flow control nozzle.		(3)
		Fuel leak into hot gas (shell) side of cooler.	Low	Increased flowrate or pressure depending upon amount of fuel decomposition.	Increased feed system pressure.	No effect	Tank gas pressure and temperature, temperature of fuel entering and leaving gas cooler, pressure and temperature at ALRC side of sonic flow control nozzle.		(3)
		External fuel leakage	Low	Slight fuel loss.	Possible fire hazard	Possible loss of mission due to aft compartment fire.	Fire detection system.		(1)
		Hot gas leakage or restricted flow	Low	Low gas pressurant flow and pressure from one engine.	Possible 20% reduction in fuel tank gas pressure.	No effect	Tank gas pressure, pressure and temperature at ALRC side of sonic flow control nozzle.	20% reduction in tank pressure will not seriously effect mission.	(3)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM BOOSTER MAIN PROPULSION PAGE 43 OF 62

SUBSYSTEM MAIN PRESSURIZATION

COMPONENT SONIC NOZZLE

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
1.3.2.2 Sonic Nozzle	Flow control of fuel tank pressurant.	Oversized	Low	Degraded control of fuel tank pressurant mass flow rate.	Increased feed system pressure.	No effect or off nominal performance (A 20% increase in tank gas pressure will not cause a tank failure.)	Pressure and temperature measurement at sonic flow control nozzle inlet. Presently baselined.		(3)
		Left out - parts will not mate at MMC/ALRC interface	Zero						-
		Restricted flow	Low	Degraded autogenous pressure	Lowered tank gas pressure.	No effect	Pressure and temperature measurement at sonic flow control nozzle inlet. Presently baselined.		(3)

SYSTEM MAIN PROPULSION PAGE 44
SUBSYSTEM MAIN PRESSURIZATION OF 62
COMPONENT BURST DIAPHRAGM (ALRC)

FAILURE MODE AND EFFECTS ANALYSIS

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
1.3.2.3 Burst Diaphragm (ALRC)	Seals fuel pressurization assembly prior to operation and facilitates system leak checks.	Premature rupture	Low	Possible solid start cartridge exhaust products introduced into pressurization system.	No effect	No effect	None	Diaphragm is coined on both sides to insure against improper installation. Parts are not interchangeable with standard burst disc assemblies.	(3)
		Fail to rupture	Low	Inability to pressurize fuel tank from one subassembly	20% reduction in tank gas pressure. (Tolerable for overall system operation.)	Possible reduction in performance.	Fuel tank gas pressure.		(3)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM MAIN PROPULSION PAGE 45
SUBSYSTEM MAIN PRESSURIZATION OF 62
COMPONENT CHECK VALVE

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
1.3.2.4 Check Valve (Fuel)	Isolates the directions of gas flow from the engine to the tank. Required only in event of engine out condition	Fails to open	Low	20% loss in fuel tank gas pressure	Possible low performance.	No effect, 20% decrease in tank pressure is tolerated by system.	Tank pressure low, pressure at sonic nozzle inlet.		(3)
		Fails to close	Low	Fuel tank gas flows back through engine after shutdown.	Loss of fuel tank gas pressure.	No effect under normal operation	Tank pressure decay	Need only operate in the event of engine out condition.	(3)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM MAIN PROPULSION PAGE 46

SUBSYSTEM MAIN PRESSURIZATION OF 62

COMPONENT FUEL PRESSURIZATION & VENT CONNECTOR

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
1.3.2.5 Fuel Pressurization and Vent Connector	Connection for on pad tank gas pressurization and venting	Leakage after ground pressurization system disconnect (poppet seal)	Low	Loss of fuel tank gas pressurization before launch.	Out of limit system pressure, i.e., tank and suction lines.	Possible delay	Tank pressure		(3)
		Redundant cap seal leakage	Low	No effect (requires two failures - connector and cap - to effect system).	No effect	No effect	None	Leakage past redundant seals are considered within tolerable limits.	(3)

FAILURE MODE AND EFFECTS ANALYSIS

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON SUBSYSTEM	FAILURE EFFECT ON SYSTEM	FAILURE EFFECT ON MISSION	RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
1.3.2.6 Fuel Tank Burst Diaphragm (MMC)	Seals fuel pressurization assembly prior to operation and facilitates system leak checks.	Premature rupture - (check valve provides redundant feature.)	Low	No effect	No effect	No effect	None		(3)
		Fail to rupture	Zero	No fuel pressurant.	Tank gas pressure could drop to a level where engine NPSH would be insufficient, also there could be possible tank structural damage.	Possible loss of mission due to tank rupture.	Tank gas pressure, pump inlet pressure.	MMC disc could possibly be eliminated with the use of line check valves. Disc is special assembly hardware which can be installed in only one way. No substitute can be made.	-

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM SOLID ROCKET MOTOR PAGE 48
SUBSYSTEM ROCKET MOTOR OF 62
COMPONENT FORWARD CLOSURE

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
2.1.1.1 Forward Closure	Contain propellant for thrust profile control, house igniter, safe and arm device, and nose section support skirt.	Propellant debond	Low	Small change in thrust history.	No effect.	No effect.	Head end pressure, tracer salts.	Excess insulation prevents burnthrough.	(3)
		Grain crack	Low	Small change in thrust history.	No effect.	No effect.	Head end pressure, tracer salts.	Excess insulation prevents burnthrough.	(3)
		Joint Leakage	Low	Structural degradation at joint.	Possible impingement of hot gas on neighboring components.	Possible loss of mission.	Pressure rate change at motor head end, burnthrough discriminator, tracer salts, on ground leak checks of assembled motor.		(1)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM SOLID ROCKET MOTOR PAGE 49

SUBSYSTEM ROCKET MOTOR OF 62

COMPONENT SEGMENT (7)

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
2.1.1.2 Segment (7)	Contain propellant and pressure during burn.	Propellant debond	Low	Possible burnthrough	Possible loss of launch vehicle due to flame impingement or SRM breakup.	Loss of mission.	Burnthrough discriminator, tracer salts, rate-head end pressure.	Actual criticality will be determined by final vehicle and orbiter configuration. Compatibility between orbiter and booster is essential.	(1)
		Grain crack	Low	Possible burnthrough	Possible loss of launch vehicle due to flame impingemnet or SRM breakup	Loss of mission.	Possible increase in chamber pressure, tracer salts.	Actual criticality will be determined by final vehicle and orbiter configuration. Compatibility between orbiter and booster is essential.	(1)
		Joint leakage	Low	Possible flame impingement on core. Possible SRM breakup.	Possible loss of launch vehicle.	Loss of mission.	Tracer salts, burnthrough discriminator, pressure rate change at head end.	Actual criticality will be determined by final vehicle and orbiter configuration. Compatibility between orbiter and booster is essential.	(1)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM SOLID ROCKET MOTOR PAGE 50
SUBSYSTEM ROCKET MOTOR OF 62
COMPONENT AFT CLOSURE

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
2.1.1.3 Aft Closure	Contain propellant, attach point for nozzle and support skirt.	Propellant debond	Low	Small change in thrust history.	No effect	No effect	Tracer salts, rate of change of head end pressure.	Excess insulation prevents burnthrough.	(3)
		Grain crack	Low	Small change in thrust history.	No effect	No effect	Tracer salts, rate of change of head end pressure.	Excess insulation prevents burnthrough.	(3)
		Leakage between segments.	Low	Decrease in thrust.	Possible loss of launch vehicle due to flame impingement on core or SRM break up.	Possible abort, possible loss of mission.	Burnthrough discriminator, on ground leak test of assembled motor, tracer salts, rate of change of head end pressure.	Actual criticality will be determined by final vehicle and orbiter configuration. Compatibility between orbiter and booster is essential.	(1)

SYSTEM SOLID ROCKET MOTOR PAGE 51

SUBSYSTEM ROCKET MOTOR OF 62

COMPONENT NOZZLE

FAILURE MODE AND EFFECTS ANALYSIS

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
2.1.1.4 Nozzle	Contain and direct thrust producing gases, house thrust vector control valves and manifold.	External leakage between nozzle and nozzle extension.	Low	Loss of thrust vector control and possible loss of nozzle.	No effect to inability to control vehicle due to TVC hardware or nozzle loss.	Loss of mission.	Thrust vector control manifold pressure switch and/or burnthrough discriminator.		(1)
		Material degradation (loss of nozzle)	Low	Loss of thrust vector control and/or reduced thrust for one SRM.	No effect to inability to control vehicle due to TVC hardware or nozzle loss.	Loss of mission.	Thrust vector control manifold pressure switch. Pitch roll or yaw commands exceed set limits.		(1)
		Loss of nozzle extension	Low	Reduction in SRM thrust.	Reduced thrust on one SRM.	Loss of mission.	Thrust vector control manifold pressure switch. Pitch roll or yaw commands exceed set limits.	Ability of safe abort is dependent on final vehicle configuration. Criticality is today's best estimate.	(2)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM SOLID ROCKET MOTOR PAGE 52

SUBSYSTEM ROCKET MOTOR OF 62

COMPONENT ROCKET MOTOR IGNITER

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
2.1.1.5 Rocket Motor Igniter (including safe and arm device)	Provide hot gas and pressure in sufficient quantity to ignite the solid rocket motor.	No fire (redundant electrical circuit)	Low	Fail to ignite rocket motor.	Fails to operate.	Loss of mission.	SRM head end pressure switches.	Provision for crew ground escape are needed.	(1)
(Safe and arm device)	Safe the rocket motor igniters so personnel may work in the area of the SRM.	Can't arm electrically	Low	S/A device inoperative.	No effect.	Launch delay - hold in countdown.	Position limit switches.		(3)
		Can't disarm electrically	Low	No effect	No effect.	Launch delay.	Position limit switches.	Remove all vehicle power to insure that inadvertant fire signal cannot be initiated.	(3)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM SOLID ROCKET MOTOR PAGE 53
SUBSYSTEM ROCKET MOTOR OF 62
COMPONENT THRUST TERMINATION DEVICE

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
2.1.1.6 Thrust Termination Device	To neutralize the net forward thrust of the SRM by opening two ports in the forward closure of the motor.	Failure to ignite	Low	Probable loss	Probable loss	Loss of mission	SRM head end pressure and head end pressure switches.	Require a double failure because device is only used for abort after another component has failed. Criticality is best estimate (Influenced by final design.)	(1)
		Premature ignition	Low	Probable loss	Probable loss	Loss of mission	SRM head end pressure. Add majority vote pressure switches to detect loss of Pc.	Criticality is best estimate. (Influenced by final design.)	(1)

SYSTEM SOLID ROCKET MOTOR PAGE 54
SUBSYSTEM ROCKET MOTOR OF 62
COMPONENT DESTRUCT DEVICE (ISDS)

FAILURE MODE AND EFFECTS ANALYSIS

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
2.1.1.7 Destruct Device (ISDS)	Directional explosive charges designed to cut open the SRM case and TVC thereby destroying the SRM system in the event of inadvertent separation.	Fail to operate	Low	Not destroyed	Not destroyed	Loss of mission	None	Inability to destroy could effect either crew or ground personnel.	(1)
		Premature operation	Low	Total loss	Total loss	Total loss	None		(1)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM SOLID ROCKET MOTOR PAGE 55
SUBSYSTEM ROCKET MOTOR OF 62
COMPONENT STAGING ROCKET MOTOR (18)

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
2.1.2.1 Staging Rocket Motor (18)	Provide thrust for SRM separation from core after SRM burn is complete.	Fail to ignite (one motor)	Low	Performance reduced	Slow SRM separation (redundant separation capability exists)	No effect, safe separation capability maintained.	None		(3)
		Premature ignition (one motor)	Low	Performance reduced	Possible vehicle disorientation (sufficient control authority exists to control the vehicle)	No effect by design	None		(3)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM SOLID ROCKET MOTOR PAGE 56

SUBSYSTEM ROCKET MOTOR OF 62

COMPONENT STAGING ROCKET MOTOR HOUSING

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
2.1.2.2 Staging Rocket Motor Housing	Act as attach points for the forward and aft staging rocket motors.	Collapse at staging motor ignition.	Low					Collapse would require a structural failure, therefore not analyzed.	

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM SOLID ROCKET MOTOR PAGE 57
SUBSYSTEM THRUST VECTOR CONTROL OF 62
COMPONENT INJECTANT TANK

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
2.2.1.1 Injectant Tank	Contains TVC fluid and pressurant necessary for expelling this fluid from propellant loading through SRM burnout.	Minor leakage, around flanges and seals.	Low	None, system able to tolerate minor loss of injectant.	None	None		Major leakages in excess of functional tolerances would require that a structural failure occur.	(3)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM SOLID ROCKET MOTOR PAGE 58
SUBSYSTEM THRUST VECTOR CONTROL OF 62
COMPONENT NITROGEN PRESSURIZATION VALVE

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
2.2.1.2 Nitrogen Pressurization Valve	Pressurize and vent TVC propellant tank.	Leakage (redundant seal - poppet and cap)	Low	No effect	No effect	No effect	Thrust vector control manifold pressure switches.	A leak sufficient to cause appreciable reduction in TVC pressure would require a structural failure.	(3)

SYSTEM SOLID ROCKET MOTOR PAGE 59

SUBSYSTEM THRUST VECTOR CONTROL OF 62

COMPONENT INJECTANT FILL & DRAIN VALVE

FAILURE MODE AND EFFECTS ANALYSIS

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
2.2.1.3 Injectant Fill and Drain Valve	Allow TVC injectant fluid to be loaded or off loaded from TVC tank.	Leakage (redundant poppet and cap seals)	Low	No effect	No effect	No effect	None	A leak of sufficient magnitude to affect the subsystem would require structural failure.	(3)

SYSTEM SOLID ROCKET MOTOR PAGE 60

SUBSYSTEM THRUST VECTOR CONTROL OF 62

COMPONENT INJECTANT TRANSFER TUBE

FAILURE MODE AND EFFECTS ANALYSIS

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
2.2.1.4 Injectant Transfer Tube	Transfer TVC fluid from TVC propellant tank to TVC manifold.	Minor leakage, around flanges and seals.	Low	None, system able to tolerate minor loss of injectant.	None	None		Major leakages in excess of functional tolerances would require that a structural failure occur.	(3)

FAILURE MODE AND EFFECTS ANALYSIS

COMPONENT INJECTANT VALVE

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON SUBSYSTEM	FAILURE EFFECT ON SYSTEM	FAILURE EFFECT ON MISSION	RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
2.2.2.1 Injectant Valves (24 total)	These valves are positioned around the nozzle and control the flow of injectant (N_2O_4) into the exhaust. This deflects the gases providing thrust vector control. The valves are divided into groups of 6, each group is located in a quadrant and are actuated simultaneously. Each valve is sealed by a "pyro plug" which is burned away by the exhaust gases.	Fail to open, one valve	Low	Loss of injectant flow from one valve	None, remaining valves provide adequate thrust vector control.	None	Valve position indicator		(3)
		Fails to close, one valve	Low	None, fail safe mechanism is actuated - see comments.	None	None	Valve position indicator	Each valve incorporates a positive closing mechanism which is activated if the valve response varies more than 3% from the control signal.	(3)
		Fails to respond to commands	Low	None, fail safe mechanism is actuated.	None	None	Valve postion indicator		(3)
		Leakage, internal prior to ignition	Low	None	None	None		Leakage is prevented from flowing onto nozzle by pyro seal.	(3)
		Leakage, internal during burn	Medium	None	None	None		Moderate leakage is tolerated by total system.	(3)
		External leakage	Low	None, system able to tolerate minor loss of injectant.	None	None		Major leakages in excess of functional tolerances would require that a structural failure occur.	(3)

FAILURE MODE AND EFFECTS ANALYSIS

SYSTEM SOLID ROCKET MOTOR PAGE 62
SUBSYSTEM THRUST VECTOR CONTROL OF 62
COMPONENT PYROSEAL (24)

COMPONENT IDENTIFICATION	FUNCTION	FAILURE MODE	PROB. OF OCCUR.	FAILURE EFFECT ON			RECOMENDATIONS FOR DETECTION	COMMENTS	CRIT. CATE.
				SUBSYSTEM	SYSTEM	MISSION			
2.2.2.2 Pyroseal (24)	Seal TVC fluid in storage tank, transfer tube and injectant manifold prior to SRM ignition.	Leakage (prior to SRM ignition)	Low	Loss of some TVC fluid before SRM ignition.	No effect	Possible launch delay to no effect	Visual after TVC load and pressurization	Injectant valves are actuated prior to ignition for flight control check - leaking pyroseal would be noticed.	(3)
		External leakage of exhaust gases around pyroseal penetration point	Low	None (see comments)	None	None		If any leakage should occur, it would be minor and would not propogate due to the configuration of this penetration point.	(3)